CONTRACT NO. NAS3-2533

NASA DOCUMENT NO. CR-54324
GE DOCUMENT NO. 65SD4296 (VOL 1)
GE DOCUMENT NO. 65SD4297 (VOL 2)
GE DOCUMENT NO. 65SD4298 (VOL 3)

NAVIGATOR
STUDY OF ELECTRIC PROPULSION FOR UNMANNED SCIENTIFIC MISSIONS

VOLUME 1
MISSION ANALYSIS

Prepared For
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
LEWIS RESEARCH CENTER
21000 BROOKPARK ROAD
CLEVELAND, OHIO

1 MARCH 1965

GENERAL ELECTRIC
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1 MARCH 1965

GENERAL ELECTRIC
MISSILE AND SPACE DIVISION

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1. INTRODUCTION

This Topical Report presents the results of studies performed by The General Electric Missile and Space Division during the nine-month extension of Contract NAS 3-2533, Study of Electric Propulsion for Unmanned Scientific Missions. Five reports* were issued in the original contract under the title of Research on Spacecraft and Powerplant Integration Problems.

The program was initiated by GE-MSD under contract to the NASA Lewis Research Center. The program objective is to determine requirements for the nuclear-electric power generating systems required in the NASA unmanned scientific probe missions throughout the solar system, which are beyond the capabilities of the presently envisioned chemical rocket propelled vehicles.

In the original contract, consideration was given to vehicles powered by post-SNAP-50 technology nuclear powerplants that began electric propulsion from earth orbit. In the contract extension, consideration is given to earlier powerplants with modest technology requirements that are launched to escape and beyond to reduce the trip time. Thus, the two studies combine to span a large spectrum of nuclear-electric propelled vehicle capabilities.

The results obtained in the current nine month study extension are presented in the following volumes:

- Volume 1 - The present volume encompasses the mission analyses. It describes the analytical techniques applied in the analyses; presents the vehicle and powerplant requirements in terms of trip time, power level, and payload for optimum

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*1. 63SD760, First Quarterly Report, 26 April to 26 July, 1963;
2. 63SD886, Second Quarterly Report, 26 July to 26 October, 1963;
3. 64SD505, Mission Analysis Topical Report, February 26, 1964;
4. 64SD700, Third and Fourth Quarterly Report, 26 October 1963 to 26 April, 1964; and
5. 64SD892, Spacecraft Analysis Topical Report, July 24, 1964.

orbiter and fly-by missions as accomplished by electrically propelled spacecraft; and presents the payload and trip time capabilities for chemical and chemical plus nuclear propelled spacecraft for the same missions.

- Volume 2 - Volume 2 compares first generation nuclear powerplants based upon an uprated SNAP 8 Mercury/Rankine Cycle, the Brayton Cycle, and the Potassium/ Rankine Cycle power systems. The comparison shows that within the limitations of the specified technologies, only the Potassium/Rankine system can result in a powerplant of sufficiently low weight to competitively accomplish a useful scientific mission. Payloads for the vehicles and operating modes for the powerplants are discussed.
- Volume 3 - Volume 3 relates the mission requirements described in Volume 1 to the power system/vehicle capabilities discussed in Volume 2. It thus defines those missions that can be accomplished with powerplants of both early and foreseeable technology and it compares the capabilities of nuclear-electric propelled spacecraft with those of chemical and chemical plus nuclear rocket propelled spacecraft.

The results show that there are useful scientific missions that can be accomplished more advantageously with nuclear-electric vehicles of even modest specific weights than with vehicles utilizing either all chemical or chemical plus nuclear rocket propulsion. A process of orderly development is, therefore, available whereby the early powerplants can be used for the near planet missions and the experience gained in these applications used to decrease powerplant specific weights. These improvements will provide powerplants of less than 30 pounds per KWe as required for more difficult planetary exploration.

2. SUMMARY

The initial phase of the mission studies performed under the subject contract was concerned with the capabilities of post-SNAP-50 powerplant technology. These studies involved the investigation of planetary orbiter missions to each of the planets of the solar system except Mars and Venus, a solar probe, and an out-of-the-ecliptic mission. Payload requirements for providing planetary and satellite soft landing capsules, high resolution radar television, and a number of sophisticated scientific experiments were identified and assumed for each of the NAVIGATOR missions. These studies were limited to the use of a single chemical propulsion stage beyond orbit and, in general, used a propulsion-coast-propulsion profile for the nuclear-electric phase of each mission. Although the results illustrated the suitability of a 1 mw powerplant for most of the NAVIGATOR missions investigated, propulsion requirements ranging from 3000 to 25,000 hours were obtained along with coasting requirements up to 20,000 hours. Only three of the missions investigated could be performed within 10,000 hours of propulsion.

This second phase of the study considers a somewhat earlier powerplant technology involving powerplant specific weights up to 70 pounds per KWe and power levels of 100 to 400 kw. Planetary fly-by missions are considered in addition to the orbiter missions. The number of initial chemical propulsion stages is increased to two stages with a maximum characteristic velocity of 40,000 fps as a means for reducing both propulsion time and trip time requirements. The mission profile includes only a single continuous electrical propulsion period to eliminate the long intermediate coast period between the two periods of operation at full power as considered in the previous study. The study includes consideration of the effects of variable specific impulse operation and consideration of chemical and nuclear propulsion mission capabilities with which to compare the above nuclear-electric propulsion results.

The initial work element of this second phase of the study included the development of a set of generalized performance characteristics which can be used to obtain low thrust propulsion requirements for the heliocentric phase of any optimum fly-by or orbiter mission in the solar system. These data were used as the basis for generating a series of mission performance maps for each of the NAVIGATOR missions. These maps show the variation in mission payload capabilities for each mission as a function of total trip time and powerplant specific weight. Auxiliary parameters displayed on these maps include propulsion time, power rating, specific impulse, and rocket characteristic velocity. Comparable data is presented, for each mission, illustrating the performance capabilities of chemical and nuclear propulsion for the NAVIGATOR type missions.

Fly-by performance data is shown for the solar probe, Mercury, Asteriod, Jupiter, and Saturn missions in conjunction with the use of the Saturn IB booster. These data are based upon the use of nuclear-electric propulsion from earth orbit with no initial high-thrust orbital propulsion. The solar probe and Mercury fly-by operation is limited to a minimum ion engine specific impulse of 3000 seconds and covers a propulsion time range of 1000 to 5000 hours. Attractive payload capabilities can be obtained for the Asteriod probe and the Jupiter fly-by for the complete range of powerplant specific weights with less than 15,000 hours propulsion time. The Saturn fly-by, on the other hand, will require propulsion times in excess of 20,000 hours with powerplant specific weights of 50 pounds per kw or greater. It represents, therefore, the limiting case for application of the Saturn IB to the NAVIGATOR missions.

Performance data is repeated for the Saturn fly-by and shown for the Uranus, Neptune, and Pluto fly-bys in conjunction with the Saturn V booster and an additional one to two stages of high-thrust orbital propulsion. These data are shown for operation at 10,000 and 15,000 hours propulsion time. The trip-time requirements for these missions range from 10,000 to 38,000 hours and the optimum specific impulse from 4000 to 7500 seconds. Operation at 10 pounds per kw is omitted since the resulting power requirements are greater than 400 kw. This operational approach is use for the Out-Of-The-Ecliptic probe, although for this mission the power requirements at 10 pounds per kw are in a region of interest. At 70 pounds per kw, on the other hand, practically no payload can be obtained.

Orbiter performance data is shown for use with the Saturn V booster and one to two stages of high-thrust orbital propulsion. The minimum specific impulse of 3000 seconds is used for the Mercury, Venus, and Mars orbiters. Propulsion time requirements for these missions range from 1000 to 5000 hours. An optimum specific impulse ranging from 3000 to 16,000 seconds is used for the remaining orbiter missions and a corresponding range of propulsion times from 4000 to 30,000 hours is obtained.

Investigations of the effects of variable specific impulse show a 10 per cent performance improvement for impulse variations of the order of 10 to 15 per cent for the relatively easy fly-by and orbiter missions. This improvement disappears, however, for the more difficult missions.

Payload capabilities are repeated for each of the NAVIGATOR missions for all high-thrust propulsion based upon the use of the Saturn IB, Saturn V, and Saturn V Nuclear boost vehicles. These data generally cover the same trip time regime as the nuclear-electric data.

3. NAVIGATOR MISSIONS

This study includes the complete spectrum of unmanned scientific exploration missions to all targets in the solar system beyond the range of the currently planned Mariner and Voyager programs. The missions investigated, therefore, exclude Venus and Mars fly-bys but include fly-bys for all the remaining planets, all planetary orbiters, a solar probe, and an out-of-the ecliptic probe. The specific missions investigated are summarized in Table 3-1.

The orbiter missions investigated are identical to those considered in the previous NAVIGATOR study (Reference 1). Each mission involves a terminal planetary descent propulsion phase which will result in an opportunity to fly-by the planet's major satellites before achieving a terminal low altitude circular orbit about the planet. As in the previous study, two missions have been identified for both the Jupiter and Saturn orbiter because of the extremely severe planetary descent propulsion requirements. Jupiter I terminates at the altitude of Callisto, the highest of its four major satellites, and Jupiter II at the altitude of Io, the lowest of its major satellites. Similarly, Saturn I terminates at the altitude of its only significant satellite, Titan, and Saturn II at the altitude of its inner ring. The Jupiter III mission of Reference 1 which terminates at a 50,000-mile radius (5600-mile altitude) is excluded from consideration because it exceeds the capabilities of the propulsion systems examined in this study. The orbital periods of all but Uranus, Neptune, and Pluto were assumed to be sufficiently small to permit the selection of a launch date for optimum rendezvous. The best rendezvous obtainable within the 1975 to 1985 time period was used for the remaining missions.

The orbiter mission profile consists of the following:

1. Injection into a 300 nautical mile Earth orbit with a two stage Saturn V booster.
2. High thrust chemical propulsion from orbit up to a maximum characteristic velocity of 40,000 fps involving one to two orbital propulsion stages.
3. Heliocentric coast.

Table 3-1. NAVIGATOR Mission Summary

Mission Type	Mission	Terminal Condition
Fly-by	Solar Probe	$5\ (10)^6$ Miles
	Mercury	Optimum Fly-By
	Asteroid Belt	Optimum Fly-By
	Jupiter	Optimum Fly-By
	Saturn	Optimum Fly-By
	Uranus	1975 Fly-By
	Neptune	1986 Fly-By
	Pluto	1986 Fly-By
	Out-of-the-Ecliptic	35 Degrees
Orbiter	Mercury	2,000 Miles Radius
	Venus	5,000 Miles Radius
	Mars	3,000 Miles Radius
	Jupiter I	1,170,000 Miles Radius
	Jupiter II	262,000 Miles Radius
	Saturn I	760,000 Miles Radius
	Saturn II	44,000 Miles Radius
	Uranus	20,000 Miles Radius
	Neptune	20,000 Miles Radius
	Pluto	5,000 Miles Radius

4. Terminal low acceleration propulsion to achieve planetary rendezvous and a terminal low altitude circular orbit.

This mission profile is illustrated schematically in the velocity diagram of Figure 3-1. The orbiter mission profile used in this study differs from the approach used in Reference 1 in that a single continuous low acceleration propulsion period is used after the heliocentric coast and that the initial high thrust propulsion limit of one stage and 20,000 fps has been increased to two stages and 40,000 fps characteristic velocity.

The fly-by missions are based upon the same planetary rendezvous requirements as their corresponding orbiters. The mission profile also involves a single continuous low acceleration propulsion phase. It is carried out, however, before the heliocentric coast. Two alternate mission profiles have been used. The first type uses the Saturn IB booster to establish the initial Earth orbit and continuous low acceleration propulsion until the required fly-by trajectory has been obtained. This approach has been used for the relatively easy fly-by missions such as the solar and Mercury probes. The alternate approach uses the Saturn V booster to establish Earth orbit, one to two orbital chemical stages to achieve hyperbolic excess velocity, and then continuous low acceleration propulsion to reduce the terminal heliocentric coast to a minimum. These two mission profiles are also illustrated schematically in Figure 3-1.

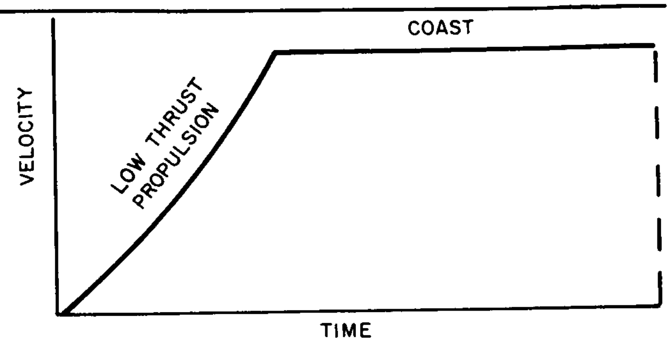
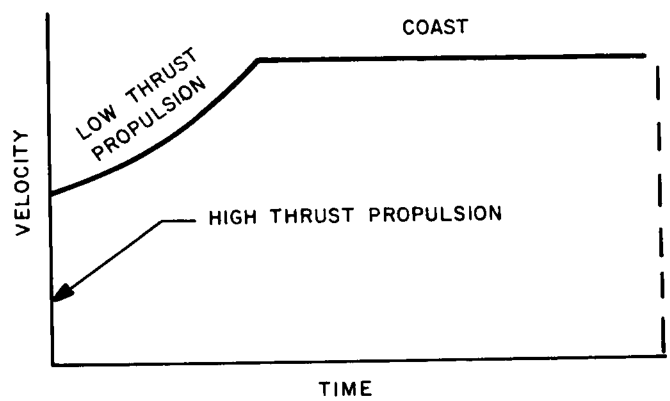
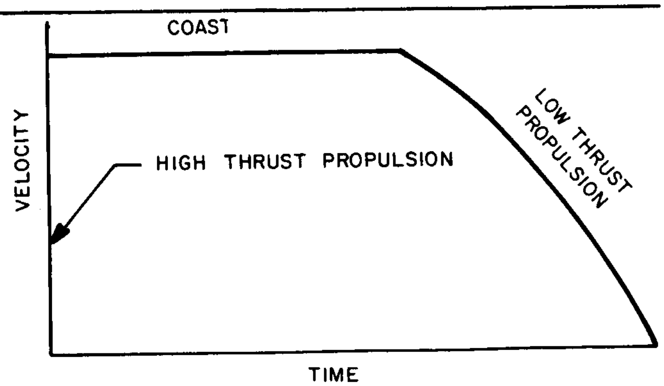
BOOSTER	MISSION	VELOCITY PROFILE
SATURN IB	FLY-BY: SOLAR PROBE TO SATURN	
SATURN V	FLY-BY: SATURN TO PLUTO	
SATURN V	ALL ORBITERS	

Figure 3-1. NAVIGATOR Mission Profile

4. TRAJECTORY ANALYSIS

Trajectory studies were conducted to develop a mathematical model of the NAVIGATOR trajectory requirements for combination with auxiliary system optimization techniques (described in Section 6.1) for use in the generation of mission performance maps for each of the various NAVIGATOR missions. In order to facilitate this process, a technique was developed which serves to seperate the heliocentric trip time effect from the propulsion requirements and which correlates the propulsion requirements for different modes of thrust operation. This technique was applied to the available data on optimum low acceleration trajectory requirements and used to develop a set of generalized heliocentric trajectory requirements. These data provide a means for determining optimum low acceleration propulsion requirements for any mission in the solar system. The results of these studies are summarized in the NAVIGATOR trajectory model.

4.1 CORRELATION TECHNIQUE; CHARACTERISTIC LENGTH CONCEPT

Optimum low acceleration heliocentric propulsion requirements can, in general, be obtained by using a computer calculation procedure based upon the calculus of variations as described in Reference 2. This approach requires an iterative solution of a two-point boundary value problem in order to achieve a specified terminal orbit and to satisfy a number of optimality conditions. A separate solution is, however, required to each combination of trip time, initial acceleration, specific impulse, initial hyperbolic excess velocity, and terminal orbit. Experience to date with the rate of convergence of the boundary value iteration has been extremely disappointing and, consequently, suggests that some correlation technique be used to minimize the number of variational solutions required.

For high thrust rocket operation, the familiar parameter, characteristic velocity, has been used successfully to correlate trajectory requirements. Characteristic velocity is relatively independent of the force field in which the vehicle operates. For low thrust trajectories, this parameter, although still independent of the force field, becomes variable with the mode of thrust operation; hence, it is desirable to determine a method of correlating the various thrust modes.

C.L. Zola (Reference 5) introduced such a method which can be used in conjunction with the characteristic velocity correlation. The correlation is accomplished by using a parameter called characteristic length which is practically invariant for different thrust modes operating in the same force field at constant trip time and central angle. Characteristic length is defined as

$$L = \int V dt \tag{4.1}$$

For a vehicle operating in one-dimensional field free space, the generalized velocity diagram is as shown in Figure 4.1-1. The generalized trajectory is composed of five segments - initial impulsive acceleration, initial low thrust acceleration, coast, terminal low thrust deceleration, and terminal impulsive deceleration. The acceleration equations corresponding to this velocity profile are integrated twice to obtain characteristic length in terms of the individual trajectory and propulsion parameters. Various special cases of the generalized mission profile for both orbiter and fly-by missions can easily be obtained. These are shown in Table 4.1-1. Characteristic velocity is directly related to mass ratio in these equations by

$$\Delta V = V_j \ln \mu \tag{4.2}$$

Using the two parameters, characteristic velocity and characteristic length, one possible correlating technique proposed by Zola is outlined in Table 4.1-2 as technique A.

The impulsive characteristic velocity requirements are determined for a given mission with a specified trip time and central angle. This then defines an approximation for characteristic length for the same mission and high thrust system in rectilinear field free space. Then, assuming L to be invariant with thrust mode, the constant low thrust characteristic velocity requirements can be calculated.

In the present NAVIGATOR study, this technique could be improved since variable low thrust data were available for most of the trajectories. Constant low thrust data were available for a limited number of trajectories, but the two point boundary value iteratio converges much more rapidly for the variable thrust mode. The variable thrust data we used instead of high thrust data as reference solutions. The procedures is outlined sche matically in Table 4.1-2 as technique B. The variable thrust propulsion requirements a embodied in $J = \int a^2\, dt$ instead of characteristic velocity.

The significance of the characteristic length concept lies in the fact that the parameter appears to be virtually independent of all of the individual propulsion parameters and to b dependent only on the trip time and the terminal orbit. This is illustrated in Figure 4.1- which summarizes the characteristic length variation for a series of Jupiter orbiter mis: For an all low thrust mission, the improved accuracy gained by using variable low thrust instead of high thrust as the reference mode can easily be seen. The accuracy decreases somewhat by using variable low thrust trajectories to determine characteristic length for combined high-low thrust mission. It appears that the resulting error is sufficiently sma although additional evaluation will be required in a more detailed investigation.

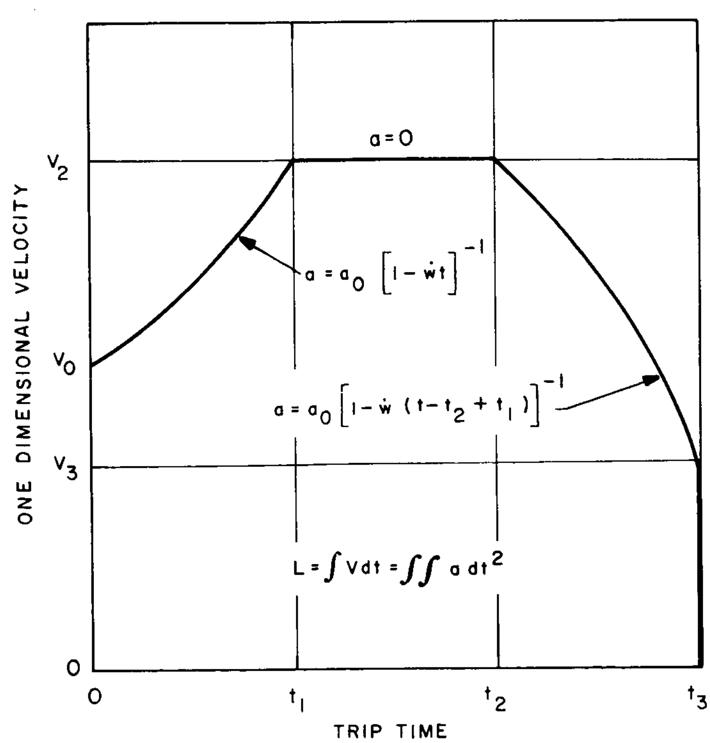


Figure 4.1-1. Generalized One-Dimensional Trajectory

Table 4.1-1. One-Dimensional Constant Thrust Trajectory Equations

Mission Profile			
Initial Propulsion	Terminal Propulsion	Velocity Equation	Characteristic Length Equation
Orbiters:			
Hi	Hi	$V_o = V_3$	$L = V_o t_h$
Hi-Lo	Hi	$\mu_1 = \mu$	$L = V_3 t_h + V_j t_p + \frac{V_j}{a_o}(V_o - V_3)$
Hi	Hi-Lo	$\mu_1 = 1$	$L = V_o t_c - (V_o - V_3) t_p + \frac{V_j}{a_o}(V_o - V_3)$
Hi-Lo	Hi-Lo	$V_3 = V_o - 2V_j \ln \mu_1 + V_j \ln \mu$	$L = V_o t_h + (V_o - V_3)\left(\frac{V_j}{a_o} - t_p\right) + \frac{V_j^2}{a_o}(1 + \mu - 2\mu_1) - V_j t_c \ln \mu_1$
Lo	Hi-Lo	$V_o = 0$	$L = -V_3\left(\frac{V_j}{a_o} - t_p\right) + \frac{V_j^2}{a_o}(1 + \mu - 2\mu_1) - V_j t_c \ln \mu_1$
Hi-Lo	Lo	$V_3 = 0$	$L = V_o\left(\frac{V_j}{a_o} + t_c\right) + \frac{V_j^2}{a_o}(1 + \mu - 2\mu_1) - V_j t_c \ln \mu_1$
Lo	Lo	$\mu_1 = \sqrt{\mu}$	$L = \frac{V_j^2}{a_o}(1 - \sqrt{\mu})^2 - 1/2\, V_j t_c \ln \mu$
Variable Lo		$V_o = V_3 = 0$	$L = \sqrt{J t_h^3 / 12}$
Fly-By:			
Hi			$L = V_o t_h$
Hi-Lo			$L = V_o t_h + V_j t_p + \left(\frac{V_j}{a_o} - t_h\right) V_j \ln \mu$
Lo			$L = V_j t_p + \left(\frac{V_j}{a_o} - t_h\right) V_j \ln \mu$
Variable Lo			$L = \sqrt{J t_h^3 / 3}$

Table 4.1-2. Thrust Correlation Techniques

Force Field	Thrust Mode	Procedure
Technique A:		
Inverse - square	High	Obtain ΔV_{hi}
Free	High	Use ΔV_{hi} to calculate L
Free	Constant low	Use L to calculate ΔV_1
Inverse - square	Constant low	Obtain propulsion requirements from ΔV_1
Technique B:		
Inverse - square	Variable low	Obtain ΔV_{v1} as represented by J
Free	Variable low	Use ΔV_{v1} to calculate L
Free	Constant low	Use L to calculate ΔV_1
Inverse - square	Constant low	Obtain propulsion requirements from ΔV_1

It is concluded, therefore, that characteristic length can be considered to be a function of only the mission and the trip time for variable low thrust, constant low thrust, and combined high-low thrust trajectories. It is further concluded that the equations of Table 4.1-1 can be used to translate the characteristic length into propulsion requirements for an assumed set of propulsion system parameters.

4.2 OPTIMUM TRAJECTORY REQUIREMENTS

The data of References 1 and 2 were used to develop the characteristic length data needed for the orbiter missions and for the fly-by missions out to Saturn. Additional optimum variable thrust trajectory calculations were obtained by the calculus of variations method for Uranus, Neptune, and Pluto fly-bys. The results of these calculations are summarized in Figure 4.2-1 along with the corresponding data on the other planets obtained from Reference 2.

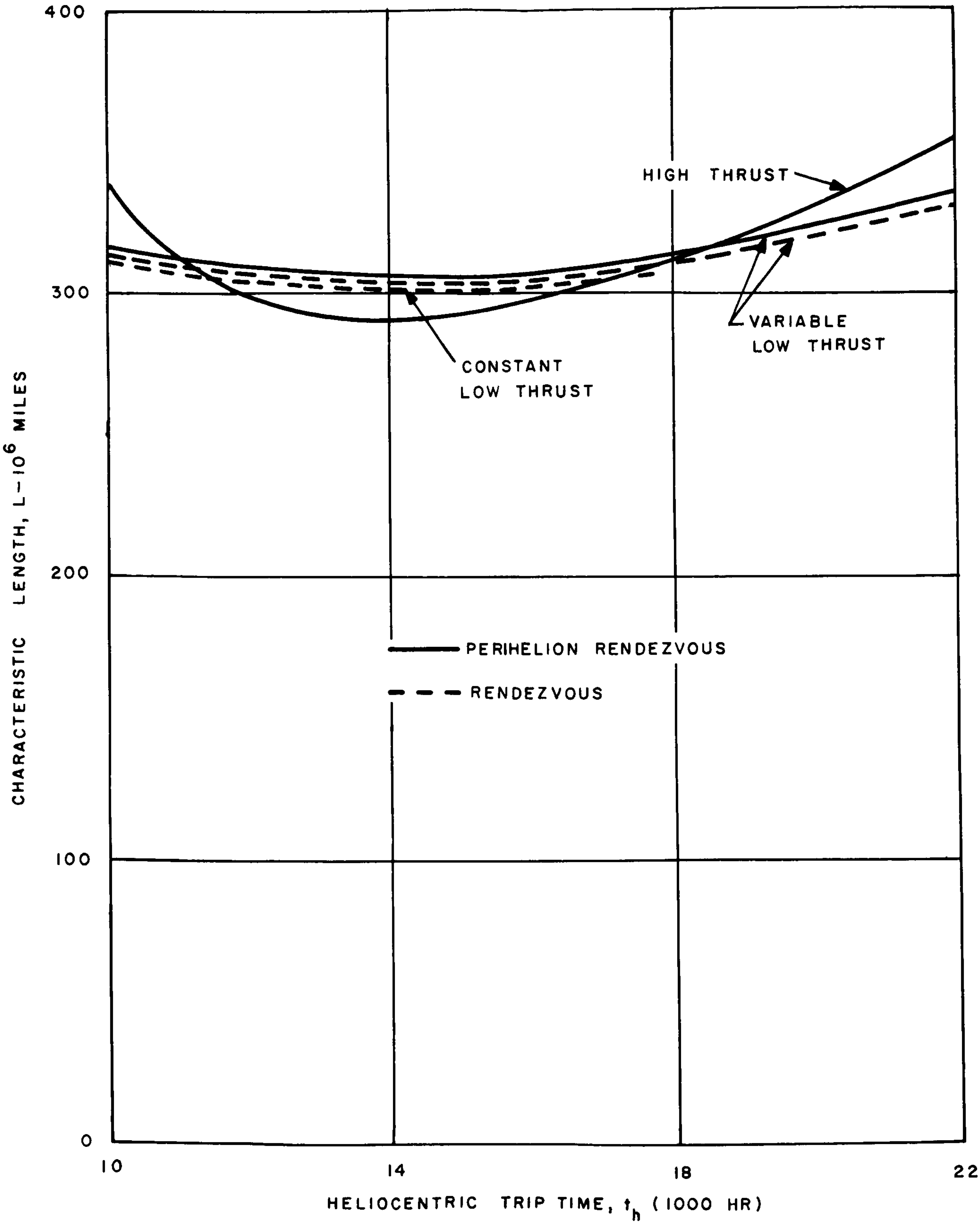


Figure 4.1-2. Characteristic Length Comparison - Jupiter Orbiter

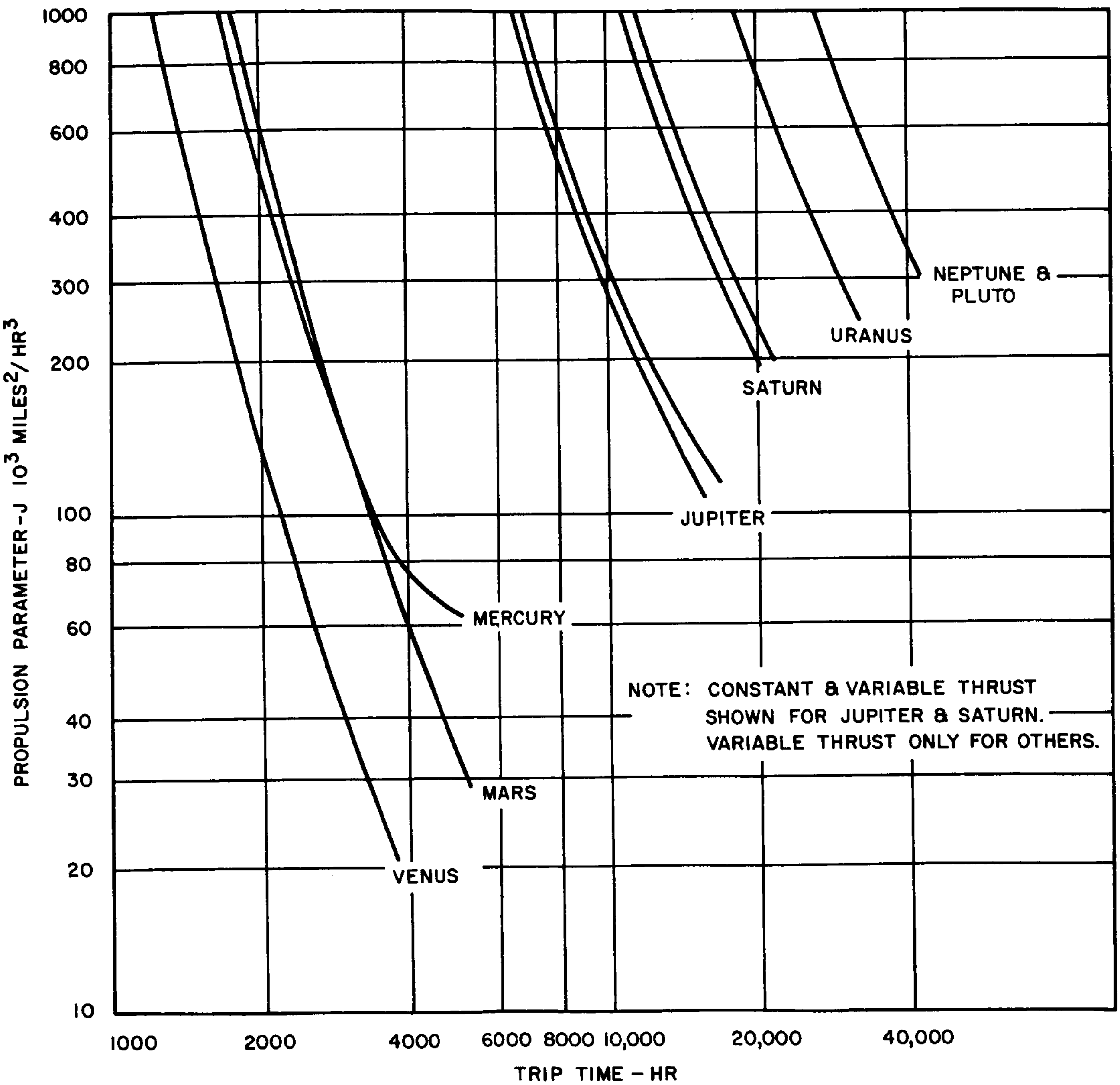


Figure 4.2-1. Fly-by Propulsion Requirements

The characteristic length was obtained from the propulsion parameter J, using the variable low thrust equations given in Table 4.1-1.

Additional constant thrust, optimum coast trajectories were calculated using the calculus of variations method; in Figure 4.2-1 these are shown as the upper Jupiter and Saturn lines. These data were used to establish additional verification of the validity of the characteristic length correlation technique. The optimum constant low thrust patterns obtained from these trajectories for typical Jupiter fly-by and orbiter missions are illustrated in Figures 4.2-2 and 4.2-3.

4.3 GENERALIZED TRAJECTORY REQUIREMENTS

Figure 4.3-1 illustrates the typical characteristic length variation with trip time obtained for each of the orbiter and fly-by missions. This type of curve can be represented empirically by a quadratic equation of the form shown in Figure 4.3-1. L_M is the minimum characteristic length and t_M is the trip time at which the minimum occurs. The third constant L_o represents an extrapolation of the characteristic length back to zero trip time and appears to have no direct physical significance.

Figure 4.3-2 summarizes the variation in t_M with terminal heliocentric distance for all of the fly-by and orbiter missions. Figure 4.3-3 summarizes the comparable variations in L_M and L_o for the outbound missions. The substantially linear characteristic of each of these curves indicate that these data will be applicable for all missions in the solar system including other solar and asteroid probes, cometary probes, etc.

Table 4.3-1 summarizes the specific values of the generalized trajectory parameters obtained for each of the NAVIGATOR orbiter and fly-by missions.

4.4 OUT-OF-THE-ECLIPTIC REQUIREMENTS

The results of the prior sections provide the necessary propulsion requirements for all of the NAVIGATOR missions except the out-of-the-ecliptic mission which requires a somewhat

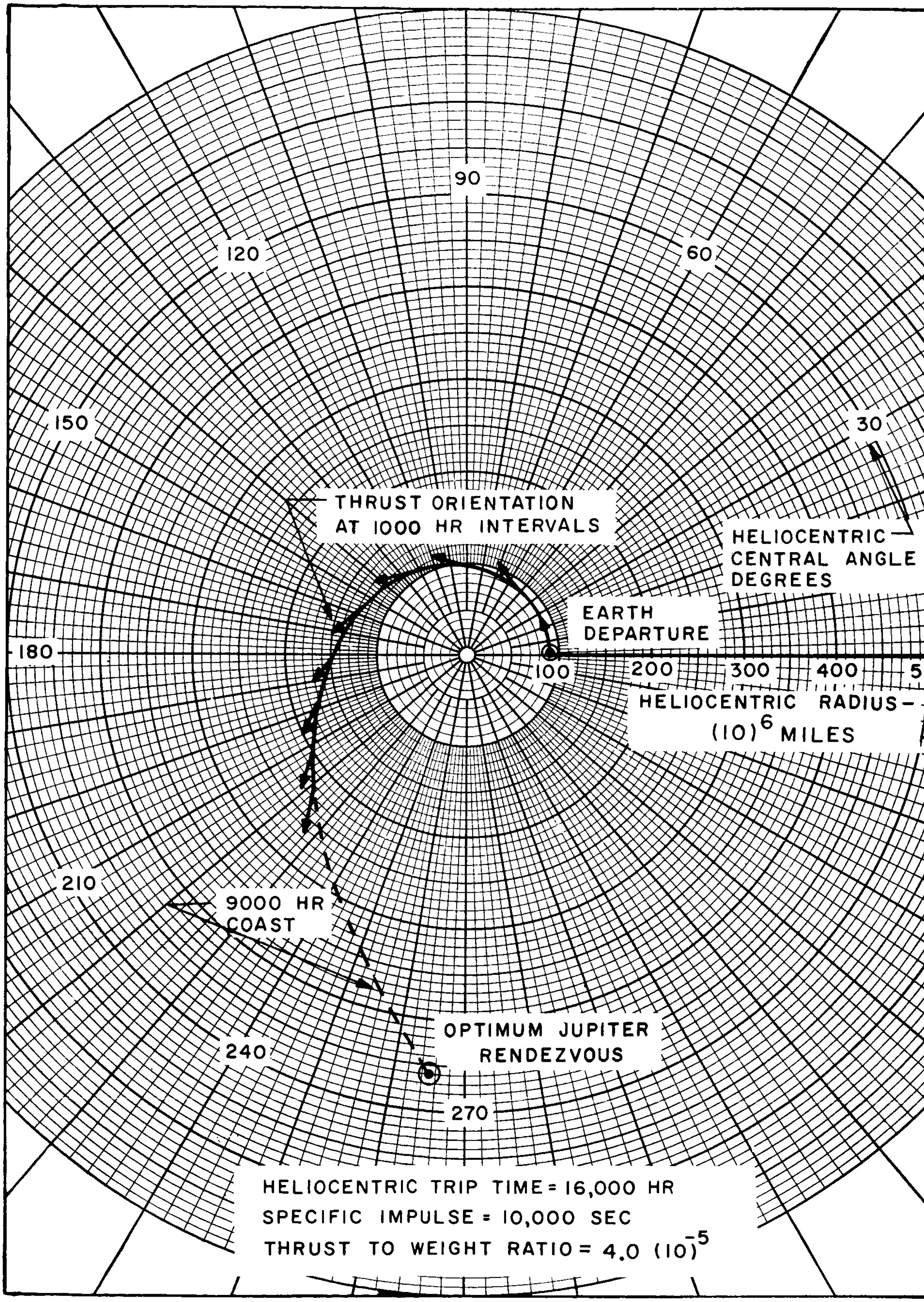


Figure 4.2-2. Jupiter Fly-by, Constant Low Thrust Pattern

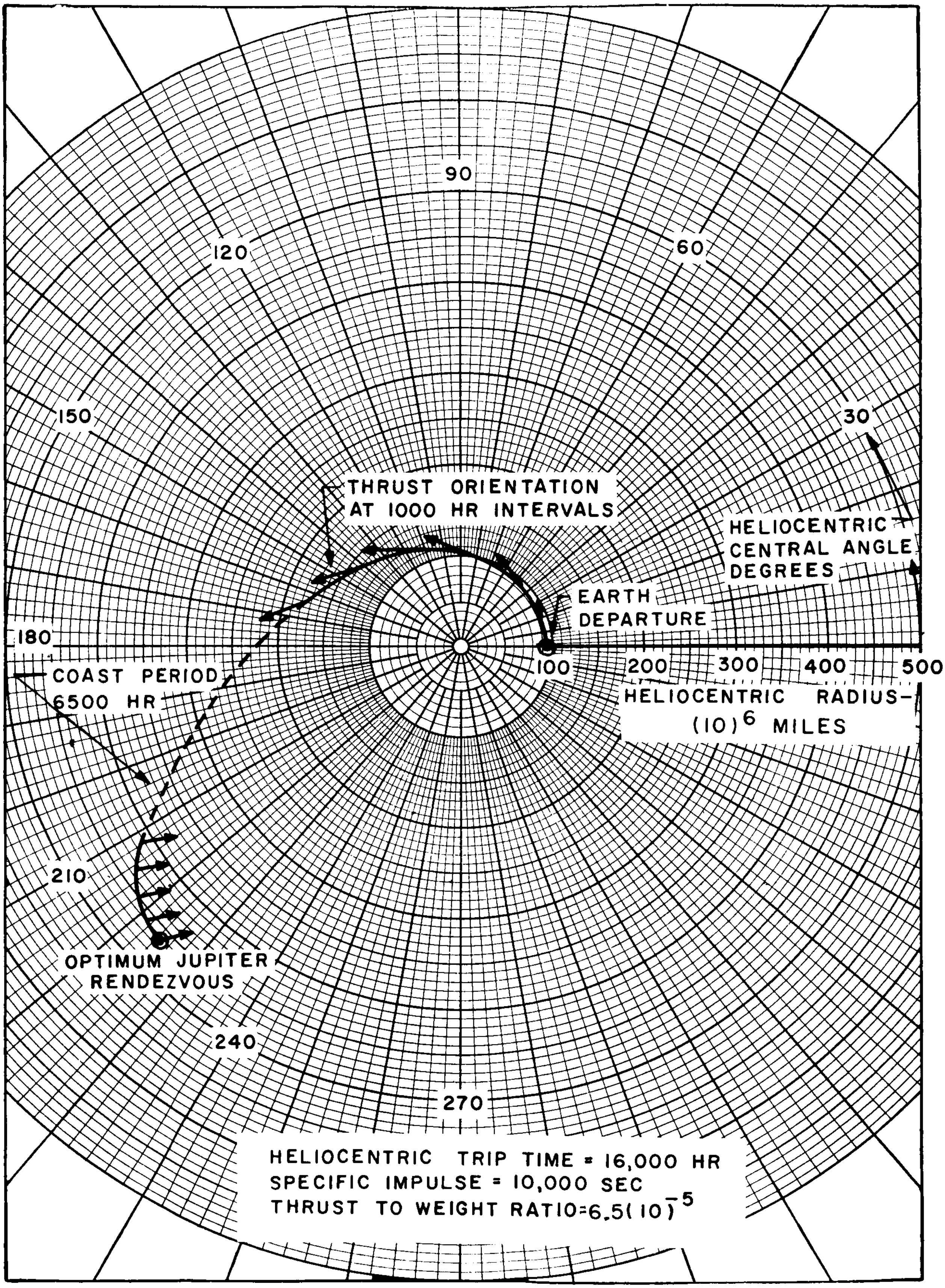


Figure 4.2-3. Jupiter Orbiter, Constant Low Thrust Pattern

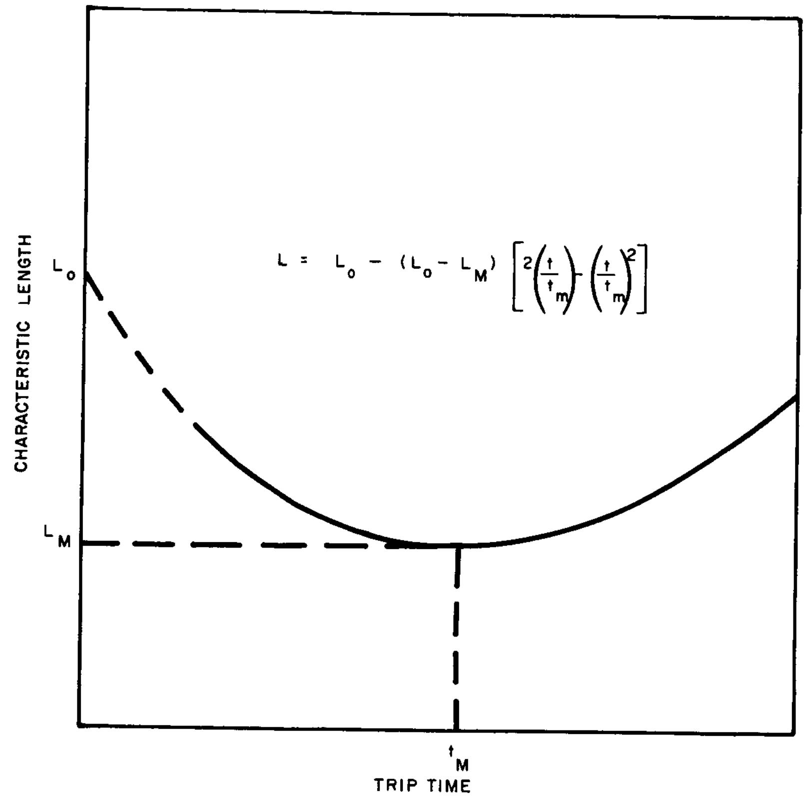


Figure 4.3-1. Characteristic Length - Trip Time Variation

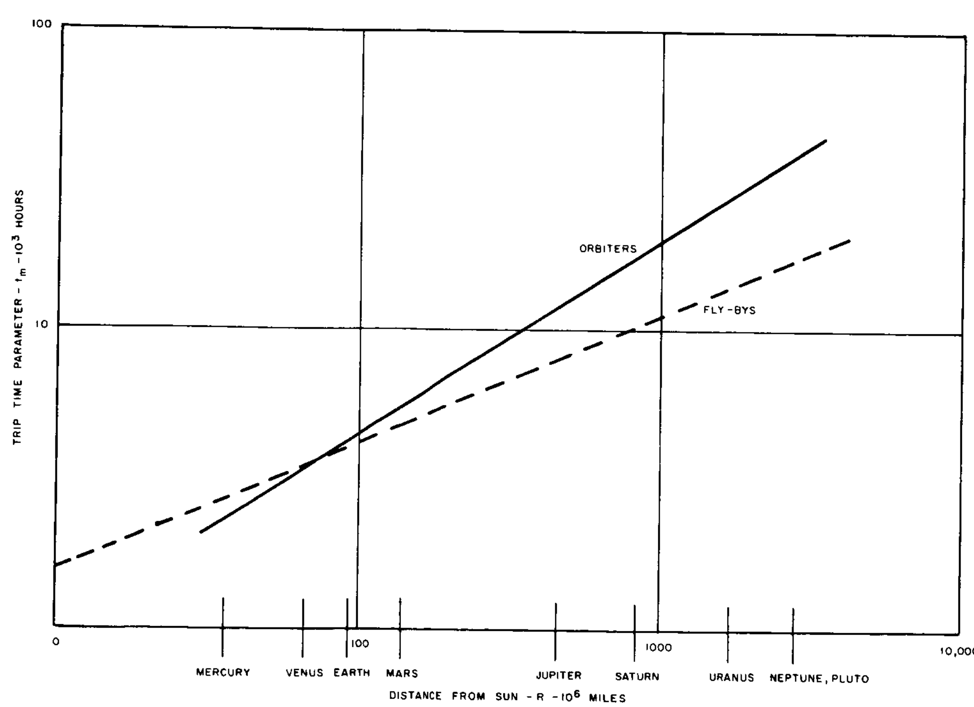


Figure 4.3-2. Correlation of Characteristic Length Parameter

Table 4.3-1. Generalized Trajectory Parameters

Mission	Type	t_M-hrs	L_M-$(10)^6$ miles	L_o-$(10)^6$ miles
Solar Probe	Fly-by	1160	2638	2650
Mercury	↓	2744	17.00	30.40
Asteroid	↓	5000	123.4	150.0
Jupiter	↓	7500	276.8	337.6
Saturn	↓	9946	589.6	679.2
Uranus	↓	14,965	1350	1541
Neptune	↓	17,454	2313	2494
Pluto	↓	17,454	2313	2494
Mercury	Orbiter	2381	39.00	57.23
Venus	↓	3452	18.49	29.00
Mars	↓	6000	27.68	46.70
Jupiter	↓	14,086	306.6	383.5
Saturn	↓	17,544	644.8	750.1
Uranus	↓	25,747	1460	1606
Neptune	↓	36,184	2519	2667
Pluto	↓	36,184	2519	2667

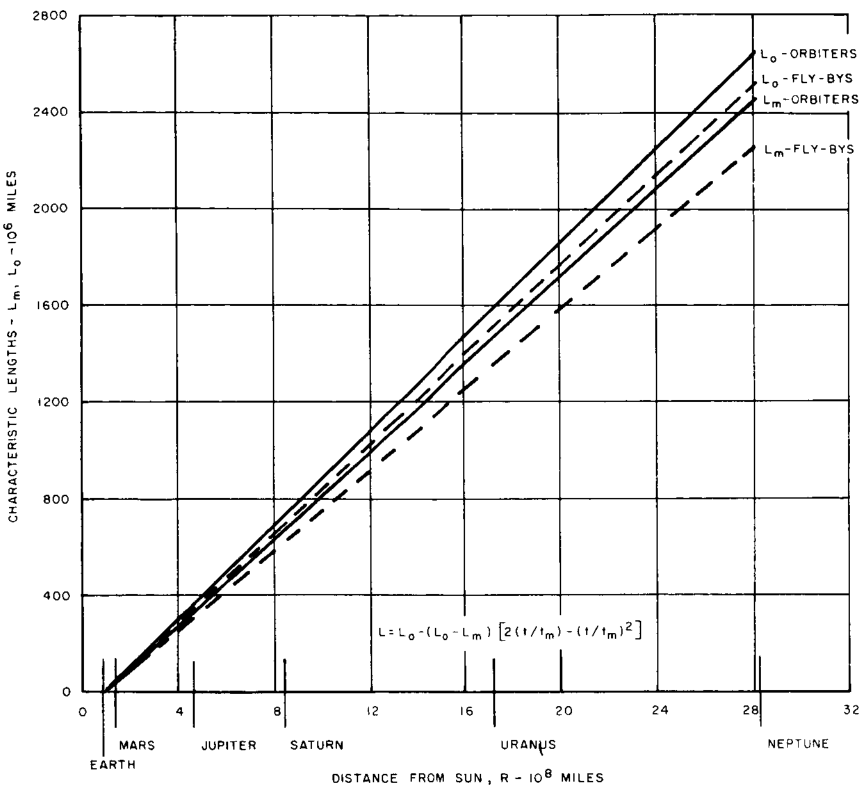


Figure 4.3-3. Correlation of Characteristic Length Parameter (Outbound Missions)

different treatment because of the three-dimensional nature of the required trajectory. Figure 4.4-1 defines the out-of-the-ecliptic thrust orientation. Figure 4.4-2 summarizes the propulsion requirements for achieving Earth satellite inclination changes for three different types of thrust orientation programs. The constant altitude case requires a constant thrust orientation angle (α) between the thrust vector and the instantaneous orbital plane of 90 degrees. Note that the sense of the thrust must be reversed every half revolution. If angle α is, however, maintained constant at 45 degrees, an in-plane thrust component is available for increasing the orbit altitude during the first half of the propulsion period and for returning the orbit altitude to its initial value by the end of the propulsion period. The higher average altitude results in a reduced average velocity which reduces the propulsion requirement for a 90-degree plane change by about 20 percent. A further reduction in propulsion requirement can be obtained by using an optimally programmed variation in α as described in Reference 4. This is indicated by the optimum angle case in Figure 4.4-2.

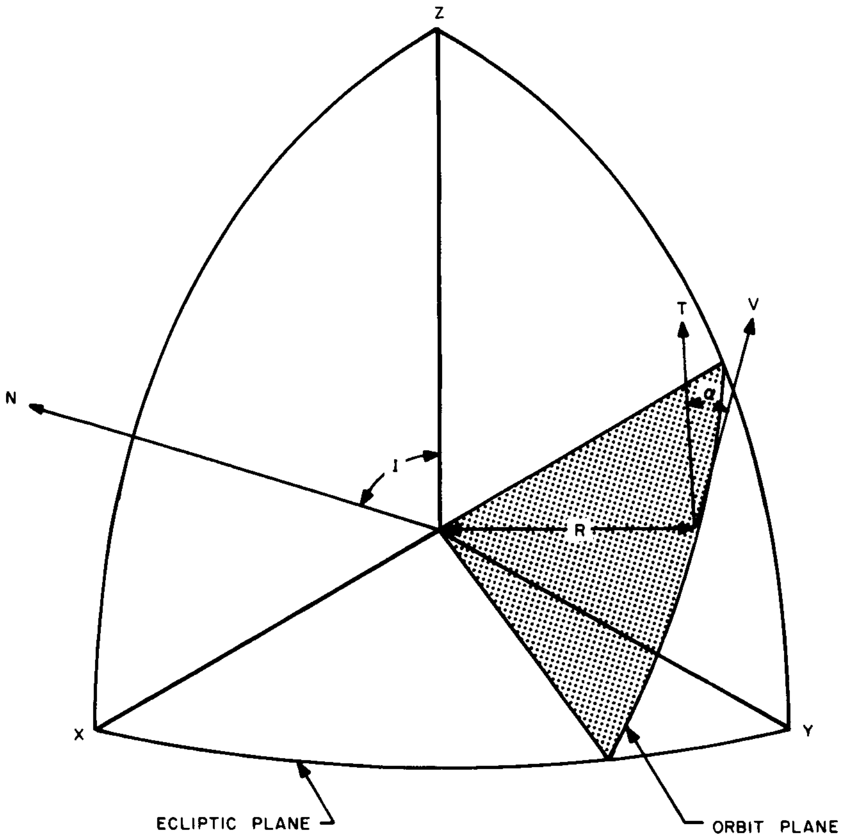


Figure 4.4-1. Out-of-the-Ecliptic Thrust Orientation

The analysis of Reference 4 is based upon an approximate quasi-circular solution which is expected to be valid for the earth satellite correction case but which is of questionable validity for the heliocentric plane change of the out-of-the-ecliptic mission. Figure 4.4-3 illustrates the propulsion requirements for the heliocentric case for the optimum and constant velocity (constant altitude) quasi-circular cases. The third curve is from actual numerical integration results for the constant velocity case. Since the optimum angle solution is essentially identical with the constant velocity solution for an inclination angle of 35 degrees, it is clear that the actual constant velocity requirement can be used. The optimum angle case might offer some reduction in propulsion requirements for inclination changes of the order of 80 to 90 degrees. This improvement has not, however, been verified by actual trajectory calculations due to a failure to obtain convergence in the two-point boundary value iteration.

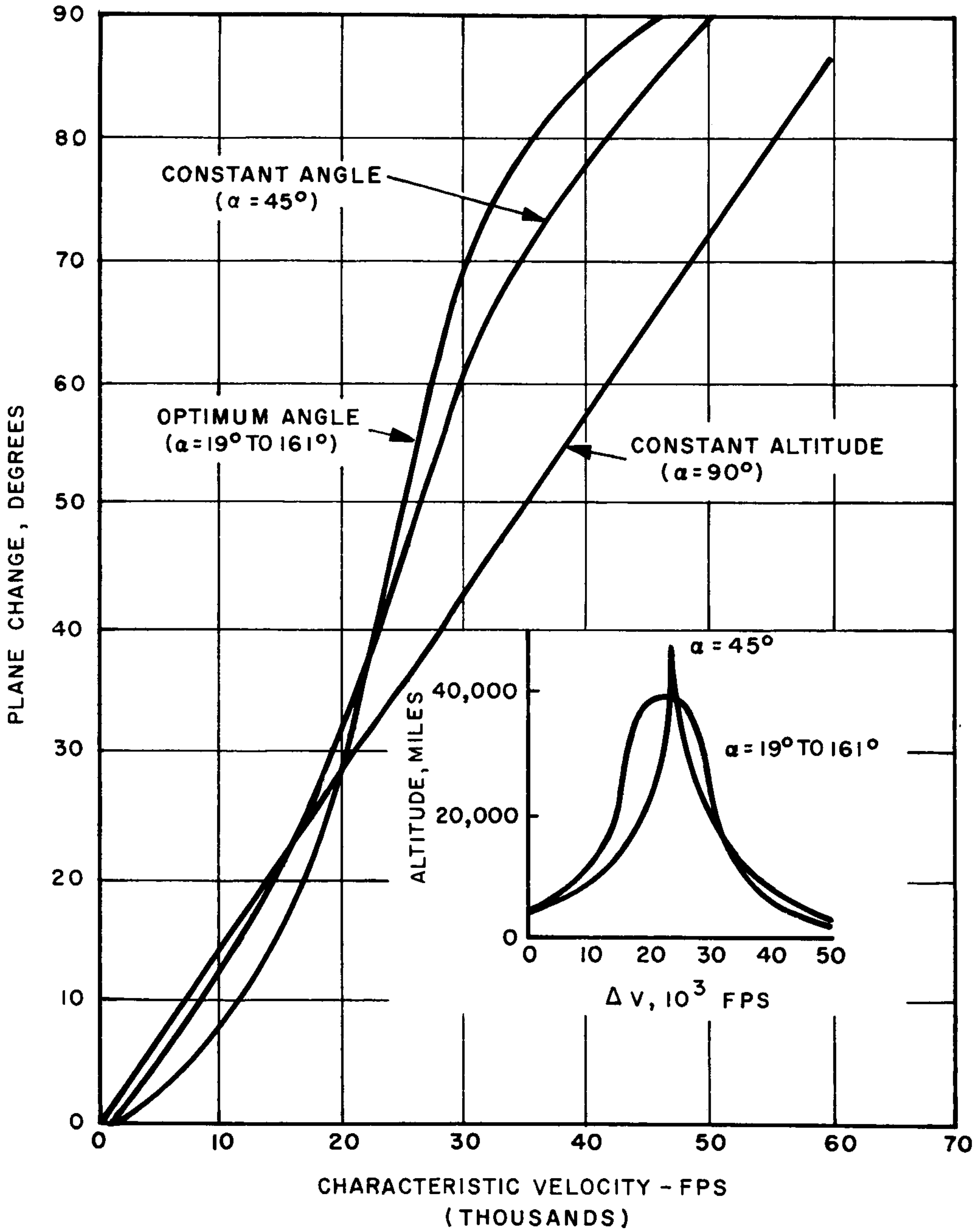


Figure 4.4-2. Low Thrust Earth Satellite Inclination Correction

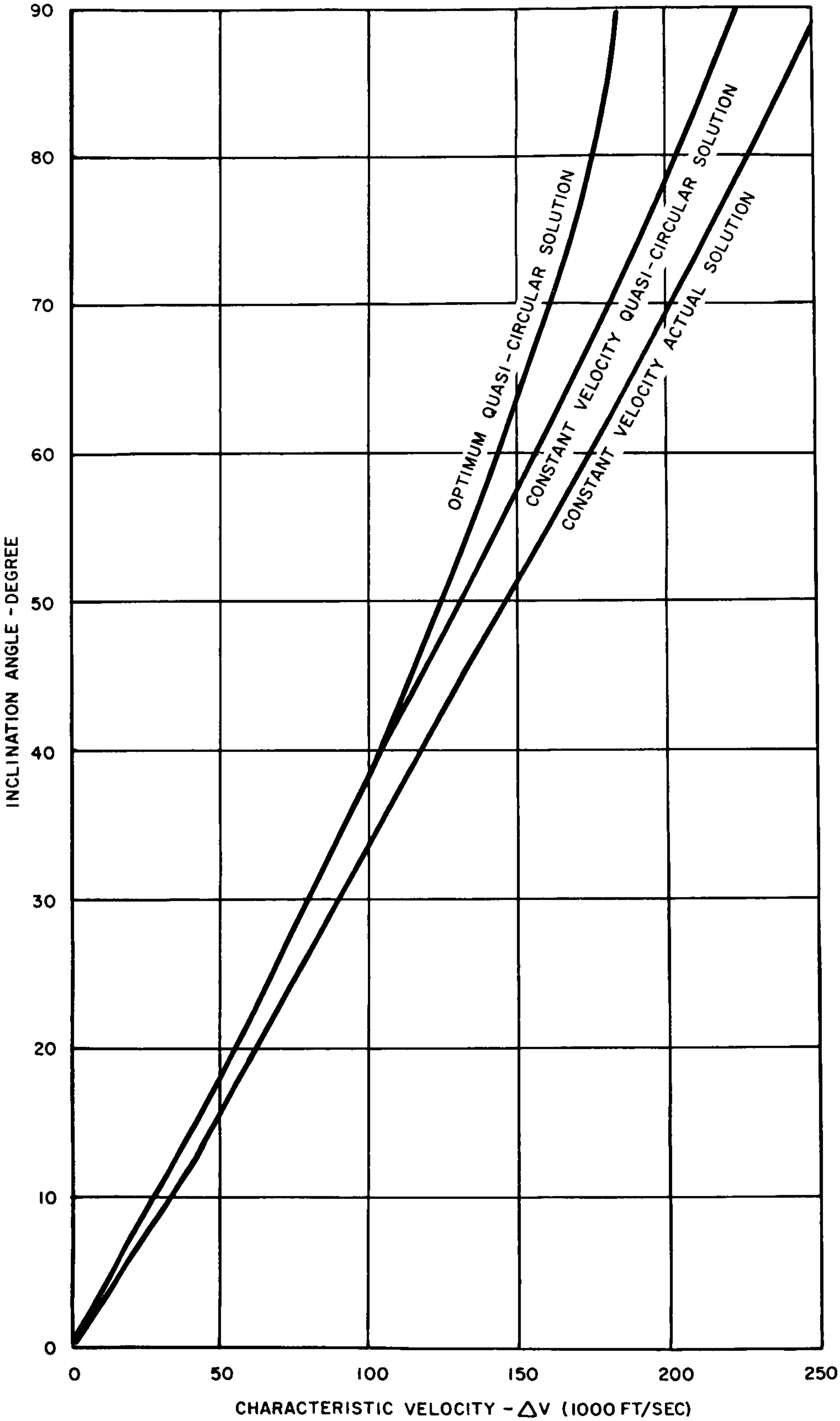


Figure 4.4-3. Out-of-the-Ecliptic Plane Change Requirements

4.5 NAVIGATOR TRAJECTORY MODEL

The results of the previous sections were used to develop the heliocentric phase of the NAVIGATOR trajectory model. The geocentric and planetary phases were obtained from Reference 1. These equations are summarized in Table 4.5-1. The second and third heliocentric equations are obtained from Table 4.1-1. The orbiter equation is the hi, hi-lo case with $V_3 = 0$. Note that the first heliocentric equation must be solved simultaneously with either the second or the third heliocentric equation to obtain the heliocentric trip time and the characteristic length. Each equation, therefore, provides the phase mass ratio as a function of the phase initial acceleration, jet velocity, propulsion time, and trip time.

Table 4.5-1. NAVIGATOR Low Thrust Trajectory Model

Phase	Mission	Equation
Geocentric	Fly-by	$\Delta V_g = \sqrt{\frac{GM_e}{P_o}} - 0.7 \left[a_o GM_e \right]^{1/4}$
Heliocentric	All	$L_o - (L_o - L_m) \left(\frac{t_t}{t_m} \right) \left[2 - \left(\frac{t_t}{t_m} \right) \right] = L$
	Fly-by	$L = V_o t_t + V_j t_p + \left[\frac{V_j}{a_o} - t_t \right] V_j \ln \mu$
	Orbiter	$L = V_o t_t - V_j t_p + \left[\frac{V_j}{a_o} - t_p \right] V_o$
Planetary	Orbiter	$\Delta V_p = \sqrt{\frac{GM_p}{P_f}} - 0.7 \left[a_o GM_p \right]^{1/4}$

5. PROPULSION SYSTEM CHARACTERISTICS

NAVIGATOR mission performance is based upon the current (1964-1965) orbital payload capabilities of the two stage Saturn IB and Saturn V boost vehicles. The initial orbit weights of these boosters is assumed to be 28,000 and 240,000 pounds, respectively. Five different upper stage configurations were identified to permit a comparison of the performance capabilities of continuous nuclear-electric propulsion with those of chemical and nuclear-rocket propulsion for the complete spectrum of NAVIGATOR missions. Table 5-1 summarizes the different vehicle configurations considered.

TABLE 5-1. NAVIGATOR VEHICLE CONFIGURATIONS

Vehicle No.	1	2	3	4	5
Stage 1	Saturn IB	Saturn IB	Saturn V	Saturn V	Saturn V
Stage 2	Saturn IB	Saturn IB	Saturn V	Saturn V	Saturn V
Stage 3	Chemical (LOX-LH)	Electrical (Hg)	Chemical (LOX-LH)	Nuclear (LH)	Chemical (LOX-LH)
Stage 4	Chemical (LOX-LH)	———	Chemical (LOX-LH)	Chemical (LOX-LH)	Chemical (LOX-LH)
Stage 5	———	———	———	———	Electrical (Hg)
Missions	Fly-By	Fly-By	Fly-By and Orbiter	Orbiter	Fly-By and Orbiter

Vehicle no. 1 involves the use of two LOX-LH chemical propulsion stages in conjunction with the basic Saturn IB vehicle. It is used to establish the reference performance capabilities of high thrust propulsion for the relatively easy fly-by missions. Vehicle no. 2, the only three stage configuration considered, uses a nuclear-electric propulsion stage in conjunction with the Saturn IB. This configuration is also utilized for the easy fly-by missions.

Vehicle no. 3 differs from no. 1 in the use of the Saturn V booster in place of the Saturn IB. It is considered for the more difficult fly-by missions and the relatively easy orbiter missions. Vehicle no. 4 differs from no. 3 in the use of a Nerva type nuclear rocket for the third stage in place of chemical propulsion. It is considered for the more difficult orbiter missions.

Vehicle no. 5 differs from no. 3 by the addition of a fifth stage using nuclear-electric propulsion. This configuration is investigated for the more difficult fly-by missions and for all of the orbiter missions.

Figure 5-1 summarizes the performance characteristics of the chemical and nuclear-rocket propulsion stages of vehicles no. 1, 3, 4, and 5. Gross payload capabilities are shown as a function of characteristic velocity from earth orbit. These data were used to define the initial nuclear-electric vehicle gross weight of vehicle no. 5 and the payload of the remaining vehicles. They are based upon the following assumptions:

1. Chemical Rocket

   Specific Impulse - 450 seconds

   Propellant Fraction (λ) - 90%

2. Nuclear Rocket

   Specific Impulse - 800 seconds

   Propellant Fraction (λ) - 88% without engine

   Shielded Engine Weight - 13,500 lb

   Engine Thrust - 50,000 lb

The performance of the nuclear-electric propulsion system is based upon the estimated capabilities of an electron-bombardment ion thruster using mercury propellant. The assumed electrical and propellant utilization efficiencies are implied by the specific power curve shown in Figure 5-2.

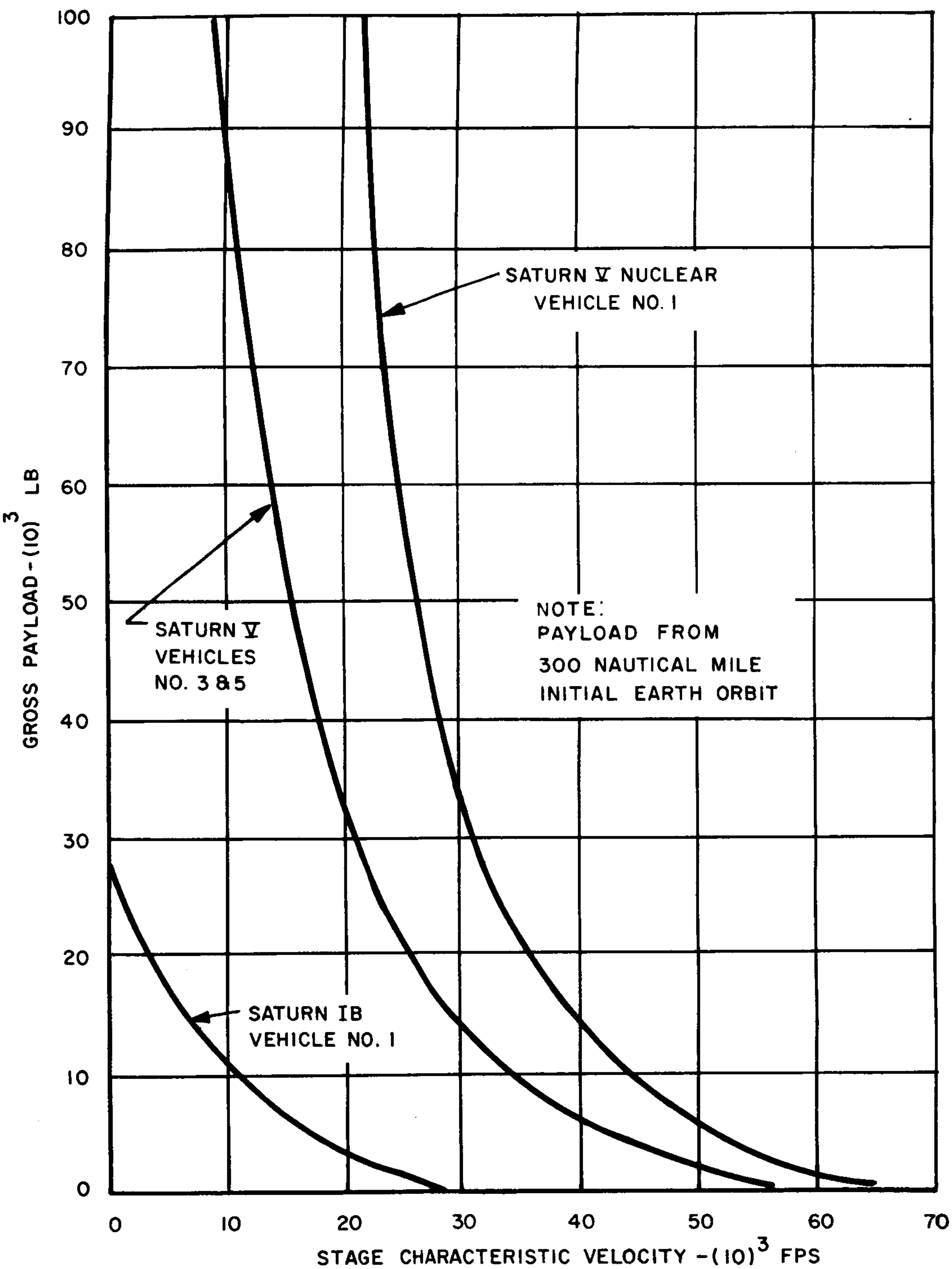


Figure 5-1. Chemical and Nuclear Rocket Performance

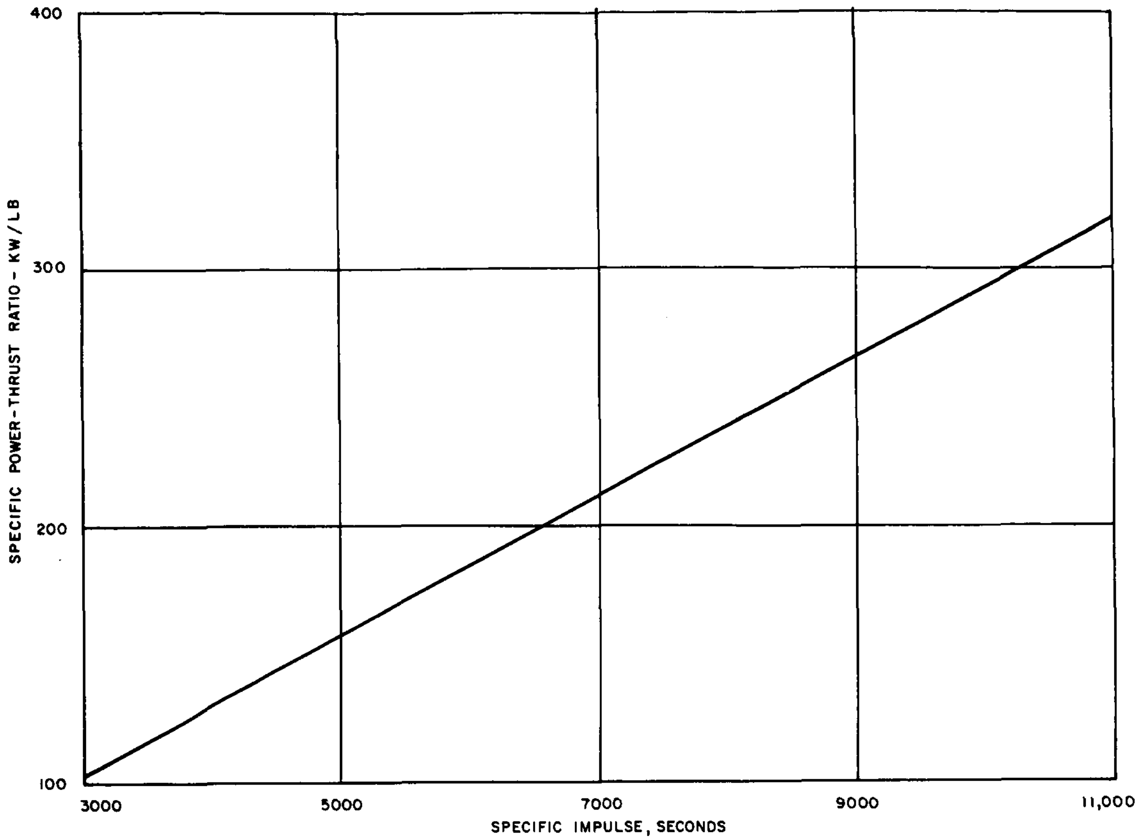


Figure 5-2. Ion Engine Performance

The linear variation with specific impulse has been represented empirically by:

$$P/T = (A_1 \; I_{sp} + A_0) \; / \; \eta_{pc}$$

where

A_1 = 0.02604 kw/lb sec

A_0 = 20.833 kw/lb

This performance is approximately eight percent lower than that used in Reference 1.

The following additional assumptions are used:

1. Propellant tankage and support fraction - 9 percent of the propellant weight.
2. Power conditioning efficiency - 96 percent.
3. Powerplant specific weight includes the weight of the powerplant power conditioning system and the electrical thrusters divided by the generator power output.

6. NAVIGATOR MISSION PERFORMANCE

The mission performance capabilities of nuclear-electric propulsion for the NAVIGATOR missions were obtained by combining the results of Sections 4 and 5. Individual mission performance, however, is a function of powerplant specific weight, the degree of initial high thrust propulsion, power level, specific impulse, propulsion time, and trip time. It is impractical, therefore, to display the performance for all possible combinations of trajectory and propulsion system design parameters. Consequently, the general problem was simplified by the identification of the optimum specific impulse and power level for achieving maximum payload at constant trip time. The results of this initial optimization process were then used to generate a series of mission performance maps for each of the NAVIGATOR missions. It must be emphasized, therefore, that the performance maps define the upper bounds on the performance capabilities of nuclear-electric propulsion and the power rating specific impulse combinations required to achieve the indicated performance. Operation is generally possible at other power specific impulse combinations but with reduced performance.

6.1 OPTIMIZATION PROCESS

The initial step in the payload optimization process involves the identification of the optimum initial high thrust acceleration as a function of the low thrust characteristic velocity and the specific impulse. This process is performed analytically and is described in the following section. The resulting equation has been included in the NAVIGATOR trajectory model and the combination used to generate preliminary working curves containing mission performance as a function of specific impulse. The working curves were then used to obtain graphical solutions for the specific impulse required for maximizing payload at constant trip time. The resulting optimum combinations of initial acceleration and specific impulse were sufficient to define the optimum power rating in conjunction with the assumed ion engine performance characteristic of Figure 5-2.

6.1.1 OPTIMUM INITIAL ACCELERATION

The general expression for the payload ratio of a nuclear-electric spacecraft can be written as

$$(W_{pl}/W_o) = 1 - (1+w_t)\,(W_{pp}/W_o) - w\,a_o\,(P/T)/_g \tag{6.1}$$

The propellant ratio can be expressed as a function of the low thrust mission characteristic velocity and the thruster specific impulse as follows:

$$(W_{pp}/W_o) = 1 - e^{-\Delta V/g\,I_{sp}} \tag{6.2}$$

The specific power can be represented by an empirical function of the specific impulse from the data of Figure 5-2 of the form

$$P/T = A_o + A_1\,I_{sp} \tag{6.3}$$

Equations 6.2 and 6.3 can then be combined with Equation 6.1 to obtain

$$(W_{pl}/W_o) = 1 - (1+w_t)\,(1 - e^{-\Delta V/g\,I_{sp}}) - \frac{w\,a_o}{g}\,(A_o + A_1\,I_{sp}) \tag{6.4}$$

Equation 6.4 can then be differentiated with respect to specific impulse with both the characteristic velocity and the initial acceleration held constant. This approach is justified by the trajectory studies of Reference 1 which indicate that both the characteristic velocity and the initial acceleration are functions only of the heliocentric trip time. This assumption is, therefore, an approximation to the desired constant total trip time case. The results of equating $\left[\partial\,(W_{pl}/W_o)/\partial\,I_{sp}\right]_{\Delta V\,\&\,a_o}$ to zero is the following equation for the initial acceleration for optimum specific impulse:

$$a_o = \frac{-(1+w_t)\,g\,u\ln\mu}{w\,A_1\,I_{sp}} \tag{6.5}$$

where the mass ratio is defined by

$$\mu = 1 - (W_{pp}/W_o) = e^{-\Delta V/g\, I_{sp}} \tag{6.6}$$

6.1.2 OPTIMUM SPECIFIC IMPULSE

The NAVIGATOR trajectory model has been augmented by Equation 6.5 and used to generate fly-by and orbiter mission performance as a function of thruster specific impulse and the other trajectory and system design parameters. A graphical process was then employed to identify the optimum specific impulse requirements. The following sections describe the specific approach used for each of the different types of missions.

6.1.2.1 Low Thrust Fly-Bys

Figure 6.1-1 illustrates the graphical process used for the fly-by missions involving no initial high thrust propulsion beyond earth orbit. These data are obtained by generating a series of data points at specific combinations of specific impulse and low thrust characteristic velocity. Constant trip time lines are superimposed on the data from a suitable cross-plot and the optimum performance line resulting in maximum payload at constant trip time identified. This line defines the optimum specific impulse-characteristic velocity relationship to be used for the final fly-by performance curves. This process was then repeated for each mission-powerplant specific weight combination.

6.1.2.2 High Thrust-Low Thrust Fly-Bys

A slightly different process is used for the fly-by missions involving initial high thrust propulsion. This configuration is used for the fly-by missions in which the previous approach results in propulsion time requirements in excess of the desired 10,000 to 15,000 hours. The function of the high thrust propulsion is, therefore, to maintain the propulsion time at this level and the analyses are made at constant propulsion times of both 10,000 and 15,000 hours.

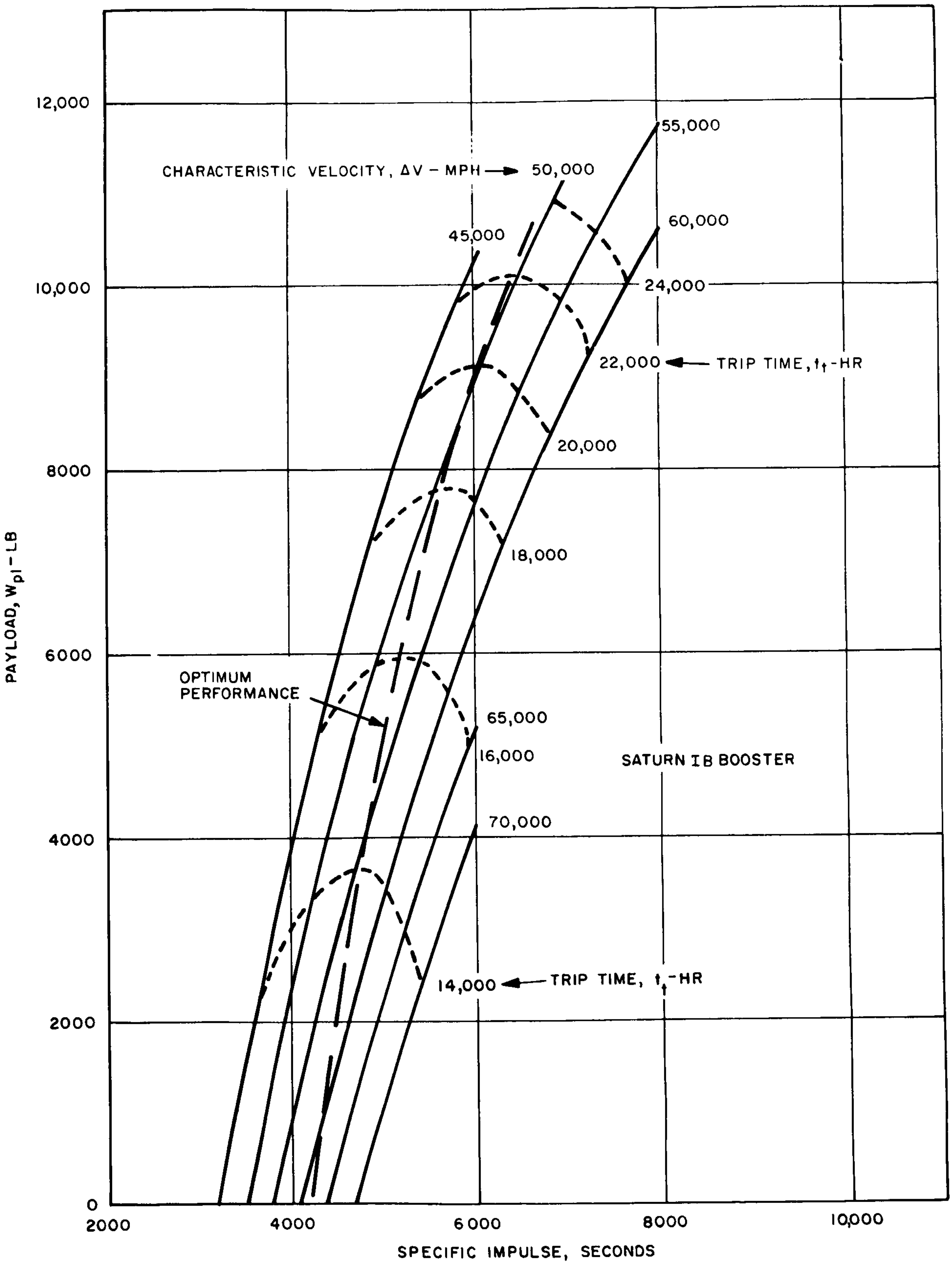


Figure 6.1-1. Jupiter Fly-By Working Curve - 50 Pounds Per KW

This approach requires the use of the alternate form of the propellant ratio equation

$$(W_{pp}/W_o) = \frac{a_o t_p}{g I_{sp}} = 1 - \mu \tag{6.7}$$

Equation 6.7 can then be combined with Equation 6.5 and solved for the specific impulse with the following result:

$$I_{sp} = \sqrt{\frac{-(1+w_t)\, t_p\, \mu \ln \mu}{(1-\mu)\, A_1\, w}} \tag{6.8}$$

Equation 6.8 is used with the trajectory model in place of Equation 6.5 for the constant propulsion time case.

Figure 6.1-2 contains a typical combined thrust fly-by working curve for constant propulsion time. These data were generated from a series of different mass ratio and rocket characteristic velocity combinations. The maximum payload envelope at constant trip time is determined and used to generate the final performance data.

6.1.2.3 Out-of-the-Ecliptic Mission

The treatment of the out-of-the-ecliptic mission differed from that of the previous section because of the fixed relationship between the low thrust characteristic velocity and the high thrust (rocket) characteristic velocity associated with a fixed inclination angle change. The use of the constant propulsion time approach will permit a direct calculation for the rocket characteristic velocity as a function of mass ratio. Equations 6.6 and 6.8 are used in conjunction with the following:

$$I_1 = f\left[\Delta V\right] \text{ - obtained from Figure 4.4-3} \tag{6.9}$$

$$I_h = I - I_1 \tag{6.10}$$

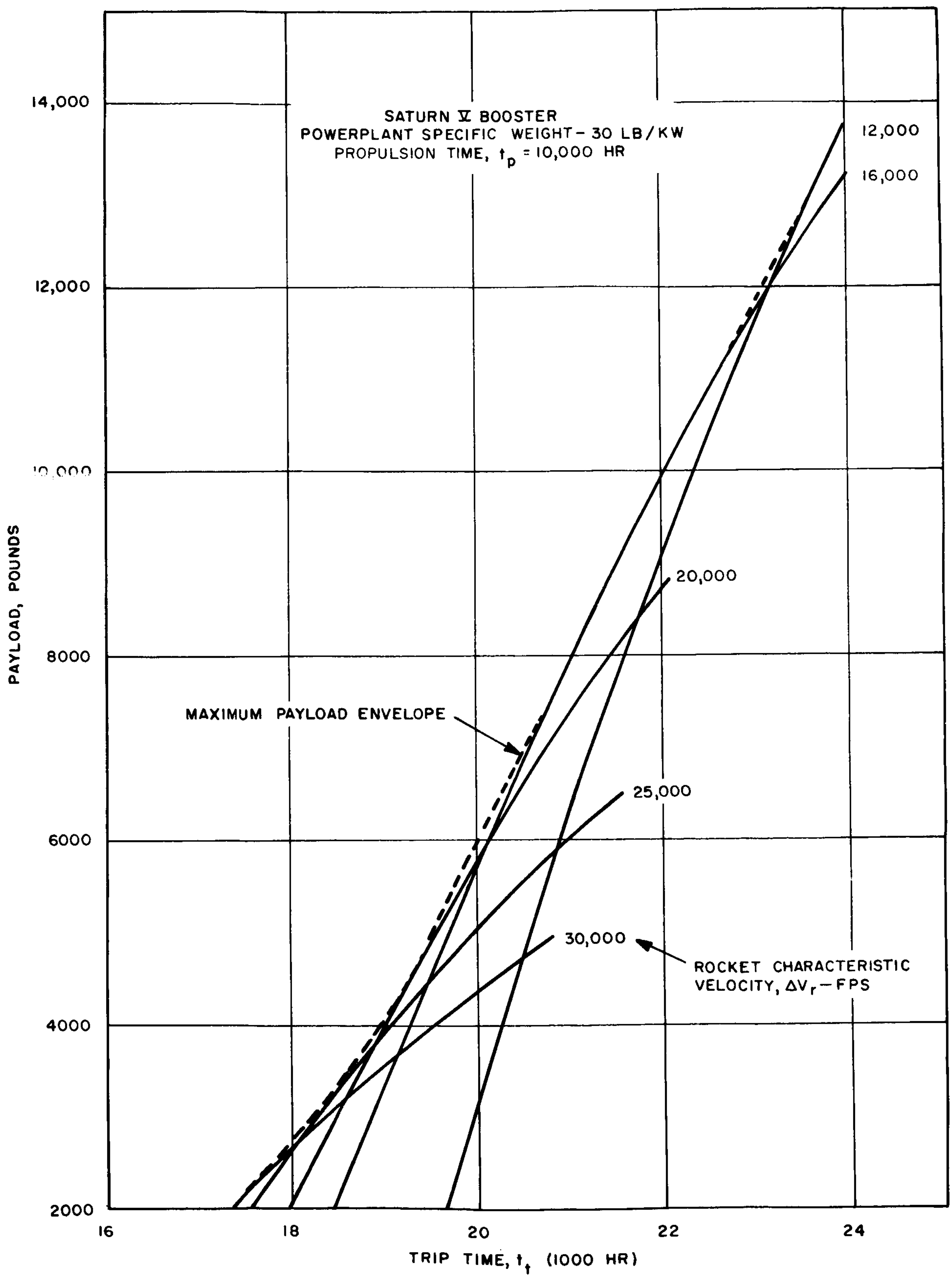


Figure 6.1-2. Uranus Fly-By Working Curve

$$V_h = 2\,V_e \sin(I_h/2) \qquad (6.11)$$

$$\Delta V_r = \sqrt{2\,V_o^{\,2} + V_h^{\,2}} - V_o \qquad (6.12)$$

Equation 6.9 is obtained from an empirical representation of the data of Figure 4.3-1.

The above approach, however, eliminates the need for the graphical optimization process since ΔV_r cannot be varied independently of the low thrust mass ratio. Consequently, the final performance data can be obtained directly from the trajectory model. These data are described in the subsequent performance section.

6.1.2.4 Orbiter Missions

The planetary rendezvous requirement imposed on the orbiter missions in conjunction with the requirement for continuous low thrust propulsion forces the hyperbolic excess velocity to be equal to the heliocentric part of the low thrust characteristic velocity

$$V_{h1} = \Delta V_h = \Delta V - \Delta V_p \qquad (6.13)$$

This is illustrated in Figure 3-1.

This constraint permits the generation of preliminary mission performance data as a function of rocket characteristic velocity and specific impulse as illustrated by a typical orbiter working curve of Figure 6.1-3. These data illustrate the variation of payload with total trip time for parametric values of rocket velocity and specific impulse. Optimum performance is obtained from the envelope of maximum payload at constant trip time. The resulting optimum performance line identifies the optimum specific impulse to be used with each specific combination of rocket velocity and powerplant specific weight.

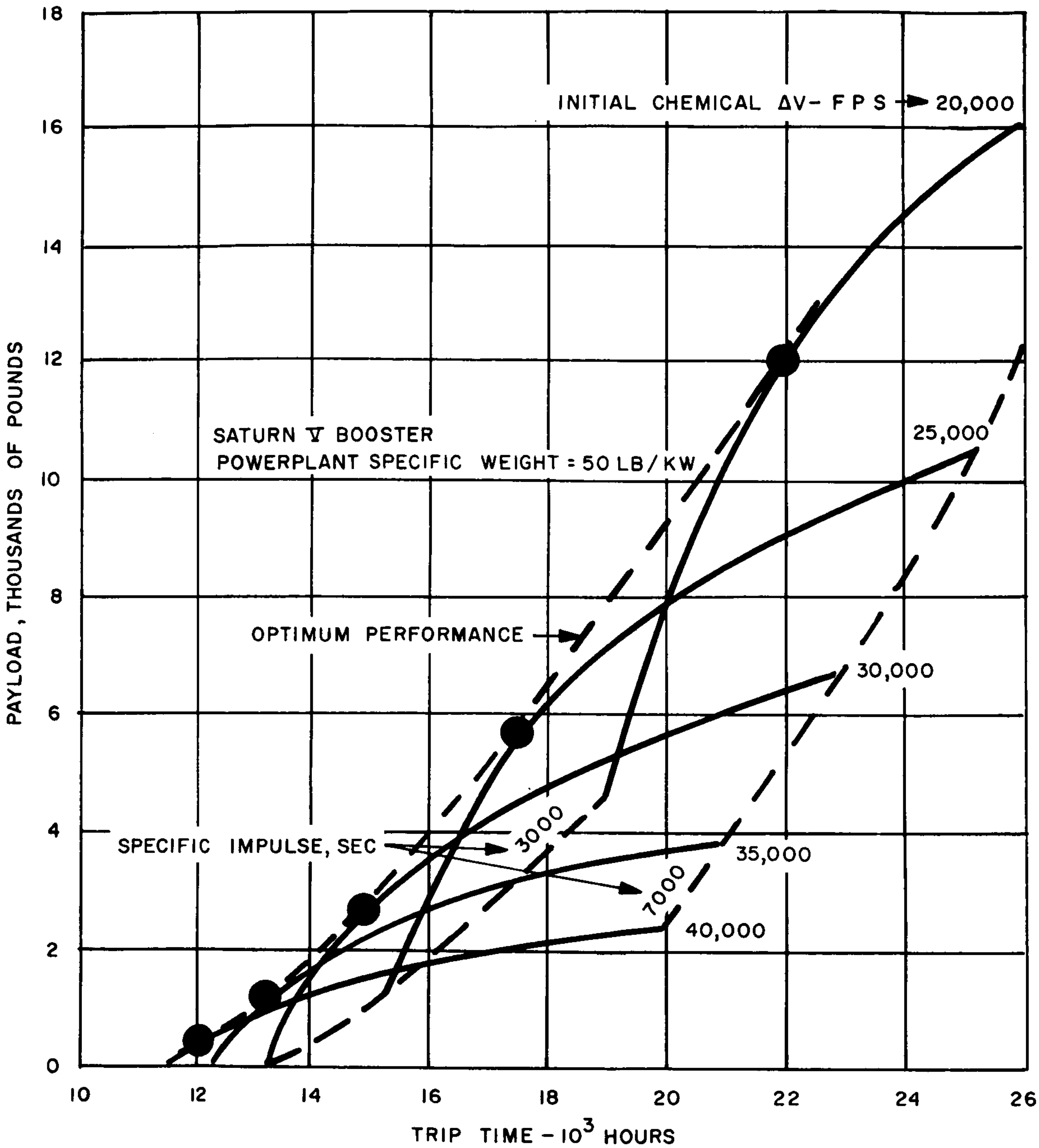


Figure 6.1-3. Jupiter I Orbiter Working Curve

6.1.3 OPTIMUM POWER RATING

The results of the previous sections define the optimum initial acceleration and the optimum specific impulse combination for achieving maximum payload at constant trip time for each of the different types of NAVIGATOR missions. These results can then be used to determine the optimum power rating from the relationship

$$P = \frac{a_o W_o}{g} \left(A_o + A_1 I_{sp}\right) \tag{6.14}$$

6.1.4 OPTIMIZATION SUMMARY

Table 6.1-1 summarizes all of the equations used in the optimization processes described in the foregoing paragraphs.

6.2 RESULTS OF MISSION ANALYSIS

The results of the optimization process are used to generate summary performance maps for each of the NAVIGATOR missions. These maps illustrate the variation in payload capabilities with trip time and powerplant specific weight. Auxiliary parameters also shown include propulsion time, power rating, specific impulse, and the Saturn V rocket characteristic velocity when applicable.

6.2.1 FLY-BY MISSIONS

The performance maps for the fly-by missions which are within the capabilities of the Saturn IB boost vehicle are shown in Figures 6.2-1 through 6.2-10. These missions include the solar probe, Mercury, Jupiter, and Saturn fly-bys. Each of these missions involves the initiation of nuclear-electric propulsion directly from low Earth orbit with no initial high thrust orbital propulsion. Propulsion time requirements are in the range of 1000 to 15,000 hours.

Table 6.1-1. Performance Calculation Procedure

Low Thrust Flybys	Combined Thrust Flybys	Orbiters
Specify ΔV, $V_j = g\, I_{sp}$, w	Specify V_o, μ, w, t_p	Specify $V_o = \Delta V_h$, $V_j = g\, I_{sp}$, w
$\Delta V_g = 15{,}600$ mph $W_o = 28{,}000$ lb	$\Delta V_r = (2V_e^{\,2} + V_o^{\,2})^{0.5} - V_e$	$\Delta V_p = (GM_p/R_t)^{0.5} - 0.7\,(a_o GM_p)^{0.25}$
$\Delta V_h = \Delta V - \Delta V_g$	$W_o = f\,(\Delta V_r)$ Figure 4.1	$\Delta V_r = 2V_e^{\,2} + V_o^{\,2})^{0.5} - V_e$
$\mu_g = e^{-\Delta V_g/V_j}$	$V_j = \left[-g^2\,(1+w_t)\, t_p \mu\, \ln\mu/A_1 w\,(1-\mu)\right]^{0.5}$	$W_o = f\,(\Delta V_r)$ Figure 4.1
$\mu_h = e^{-\Delta V_h/V_j}$	$a_o = V_j\,(1-\mu)/t_p$	$\mu_h = e^{-\Delta V_h/V_j}$
$\mu = \mu_g \mu_h$		$\mu_p = e^{-\Delta V_p/V_j}$
$a_o = -g^2\,(1+w_t)\,\mu\,\ln\mu/w\,A_1\,V_j$		$\mu = \mu_h \mu_p$
$a_1 = a_o/\mu_g$		$a_o = -g^2\,(1+w_t)\,\mu\,\ln\mu/w\,A_1\,V_j$
$a = (L_o - L_m)/t_m^{\,2}$	$a = (L_o - L_m)/t_m^{\,2}$	$a = (L_o - L_m)/t_m^{\,2}$
$b = V_j \ln\mu_h + 2\,(L_m - L_o)/t_m$	$b = -V_o + V_j \ln\mu + 2\,(L_m - L_o)/t_m$	$b = -V_o + 2\,(L_m - L_o)/t_m$
$c = L_o - V_j^{\,2}\,(1 - \mu_h + \ln\mu_h)/a_1$	$c = -t_p V_j \left[\ln\mu/(1-\mu) + 1\right]$	$c = L_o - \left[V_j\,(\mu_h - 1) + V_o \mu_h\right] V_j/A_1$
←	$t_h = \left[-b - (b^2 - 4\,ac)^{0.5}\right]/2a$	→
$t_g = (1 - \mu_g)\,V_j/a_o$	$t = t_h$	$t_{p1} = (1 - \mu_p)\,V_j \mu_h/a_1$
$t = t_g + t_h$		$t_p = (1 - \mu)\,V_j/a_o$
		$t = t_h + t_{p1}$
←	$W_f = \mu W_o$; $P = W_o a_o\,(A_1 I_{sp} + A_2)/g\,\eta_{pc}$; $W_{ps} = wP$; $W_{eng} = w_e \eta_{pc} P/I_{sp}^{\,0.5}$ $W_{p1} = W_f - w_t\,(W_o - W_f) - W_{ps} - W_{eng}$	→

Figure 6.2-1 contains the payload variation with trip time for lines of constant powerplant specific weight for the solar probe mission. Optimum performance is obtained at a specific impulse of 3000 seconds, the minimum ion engine specific impulse permitted in the investigation. Note that performance at low trip times is limited by the no coast limit at which point the propulsion is continuous for the duration of the mission. The 70 pounds per kw powerplant specific weight operation cannot be obtained within this no coast limit. Propulsion time requirements are seen to be within the range of 1000 to 5000 hours. Figure 6.2-2 summarizes the associated power requirements as a function of trip time and powerplant specific weight. Note that the 70 pounds per kw operation is also lower than the 100 kw minimum power level of interest.

Figures 6.2-3 and 6.2-4 contain similar data on the Mercury fly-by mission. These data are quite similar to the preceding set. Performance is not shown for 10 pounds per kw because the associated power requirements are well beyond the 400 kw maximum power level of interest. Operation at 70 pounds per kw is possible for the Mercury mission and is included on the performance map.

Figures 6.2-5 through 6.2-10 contain the performance maps for the Asteroid, Jupiter, and Saturn fly-bys. These data differ from the preceding in that the optimum specific impulse is in excess of the 3000-second minimum. These data are characterized by propulsion time requirements which are dependent only on the trip time and by finite terminal coasting periods over the complete range of operation. Performance is generally shown for the complete 10 to 70 pounds per kw range of powerplant specific weight. The propulsion time requirements are within 15,000 hours for all of the Asteroid and Jupiter fly-by data. The Saturn fly-by, however, requires propulsion time requirements in excess of 20,000 hours for powerplant specific weights greater than 50 pounds per kw. The Saturn fly-by mission, therefore, represents the limiting case for the use of the Saturn IB boost vehicle with no initial high thrust orbital propulsion.

Figures 6.2-11 through 6.2-22 contain the fly-by performance maps based upon the use of the Saturn V boost vehicle and one to two stages of initial high thrust orbital propulsion.

The high thrust propulsion is used in all cases for maintaining the propulsion time requirements at the 10,000 and 15,000 hour levels. Saturn fly-by performance is repeated for this mode of operation in Figures 6.2-11 and 6.2-12. Figure 6.2-11 contains the Saturn fly-by operation with the Saturn V booster at 10,000 hours propulsion time. Payload is plotted against trip time, and lines of constant powerplant specific weight and rocket characteristic velocity are shown. Operation at 10 pounds per kw has been omitted due to excessive power requirements. Figure 6.2-12 contains the power variation with trip time for each powerplant specific weight. Note that the optimum specific impulse is dependent only on the powerplant specific weight. Saturn fly-by performance at 15,000 hours propulsion time is not included since the 10,000 hour performance appears to be acceptable.

Figure 6.2-13 through 6.2-16 contain comparable performance data for the Uranus fly-by for operation at both 10,000 and 15,000 hours propulsion time. These data are generally similar to the previous Saturn fly-by data. The primary effect of the increased propulsion time appears to be a slight reduction in power requirements and a significant reduction in optimum specific impulse requirements. Figures 6.2-17 through 6.2-20 contain similar data for the Neptune and Pluto fly-by missions. Note that the fly-by requirements for both planets are identical for the assumed 1985 launch date. Some variation in optimum specific impulse at constant specific weight is obtained for these missions as shown in Figures 6.2-18 and 6.2-20. Note that the trip time requirements for these fly-by missions range up to 38,000 hours.

Performance for the out-of-the-ecliptic mission is shown in Figures 6.2-21 and 6.2-22. Figure 6.2-21 contains the variation in payload with powerplant specific weight and power level for operation at 10,000 hours propulsion time. Trip-time requirements cannot be determined by a parametric study of this type but can generally be expected to exceed the propulsion time by 2000 to 4000 hours. Note that operation is extremely marginal with powerplant specific weights in excess of 30 pounds per kw. Figure 6.2-22 contains similar data for operation at 15,000 hours propulsion time. These data show substantial payload improvements of up to 50 percent at constant power.

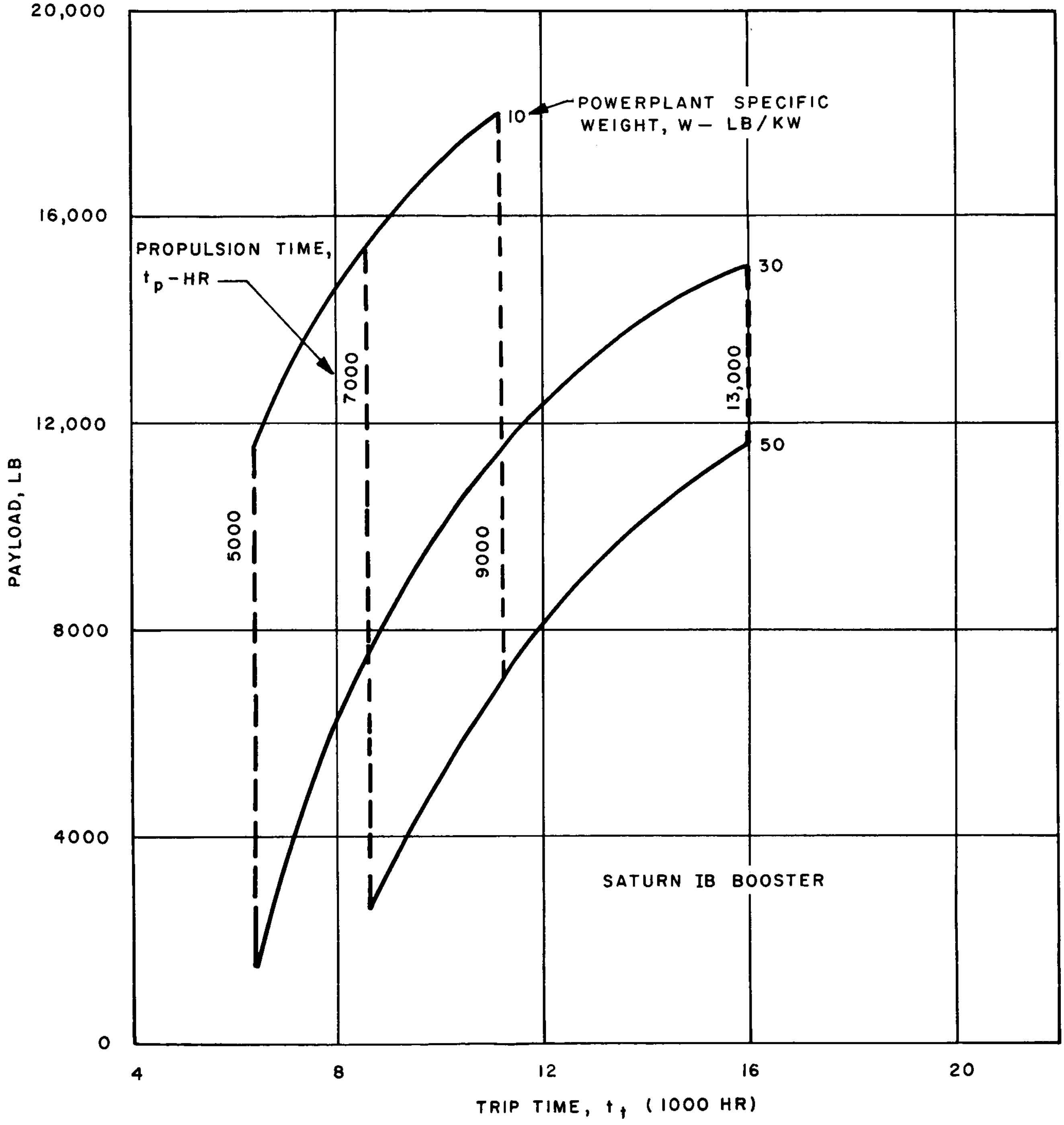


Figure 6.2-1. Solar Probe (5 x 10^6 Mile) Performance

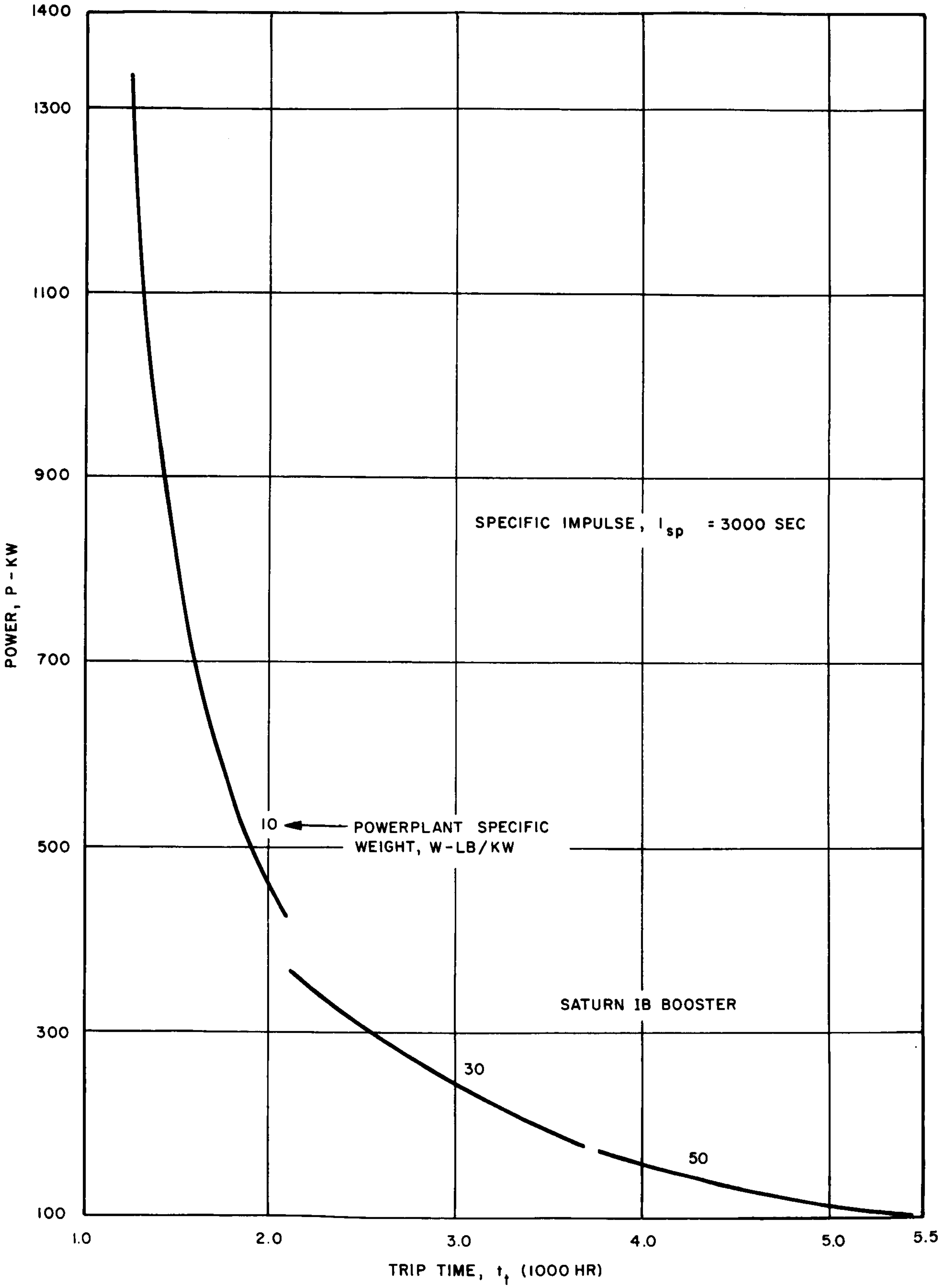


Figure 6.2-2. Solar Probe (5 x 10^6 Mile) Requirements

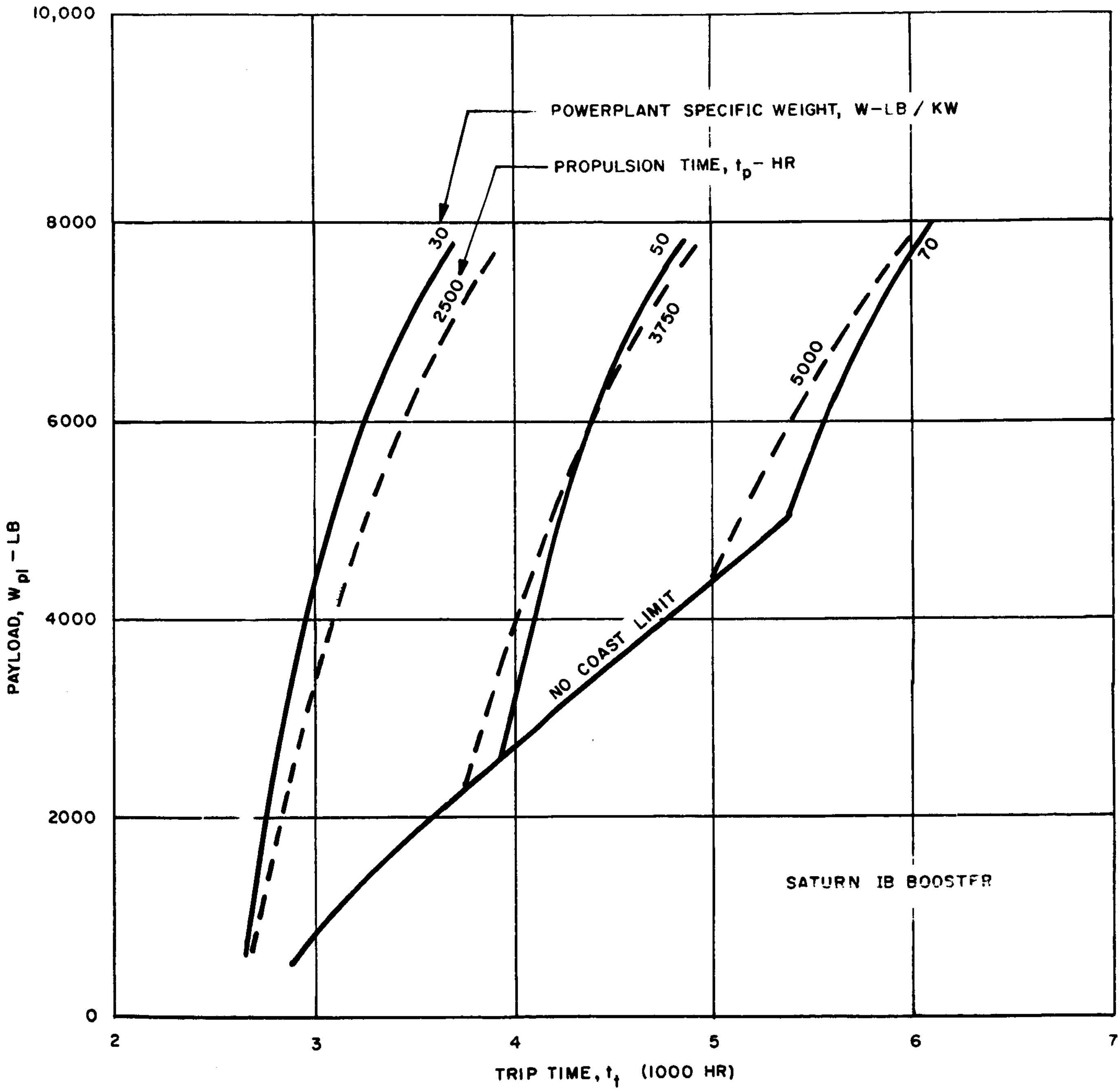


Figure 6.2-3. Mercury Fly-By Performance

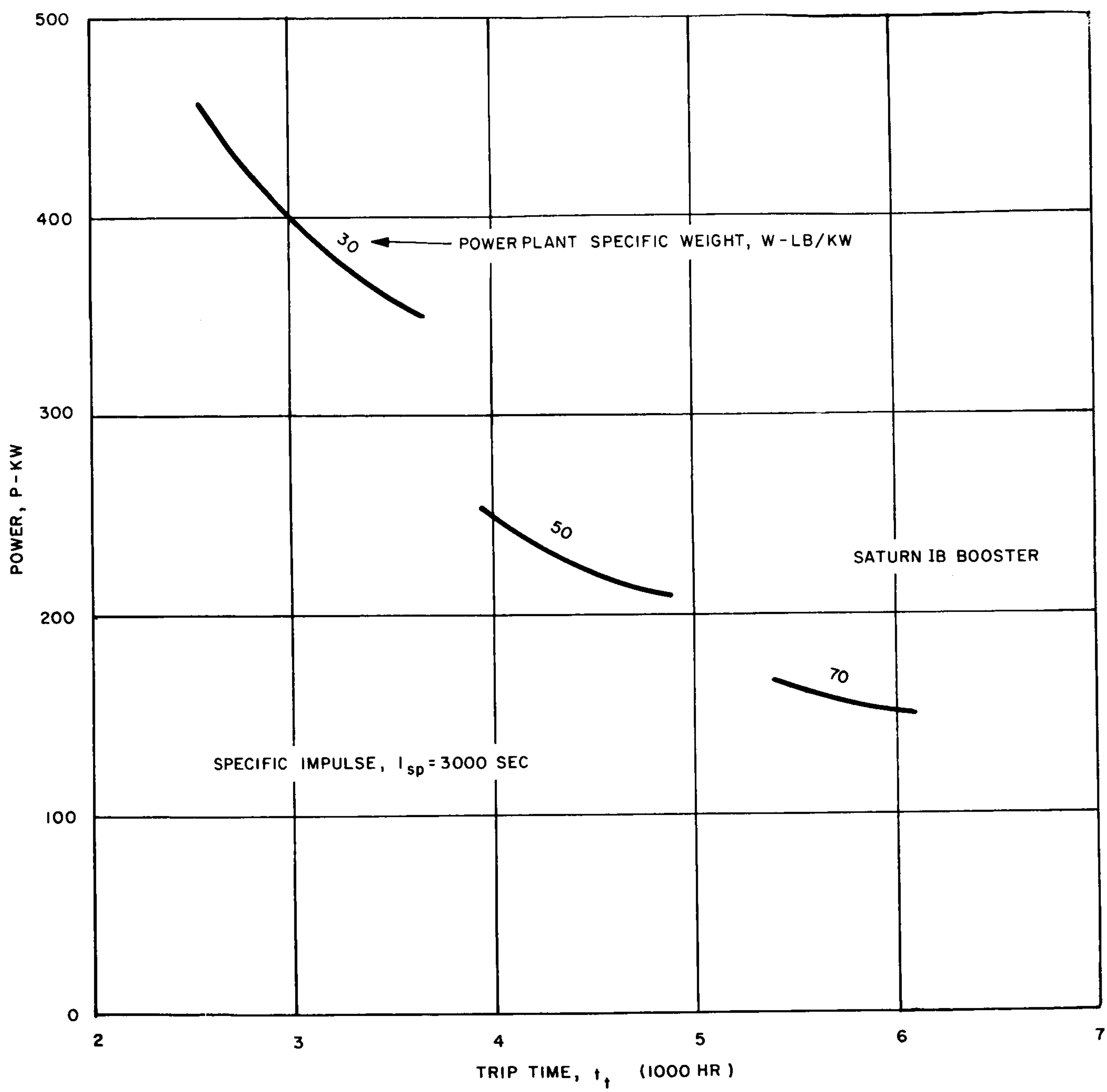


Figure 6.2-4. Mercury Fly-By Requirements

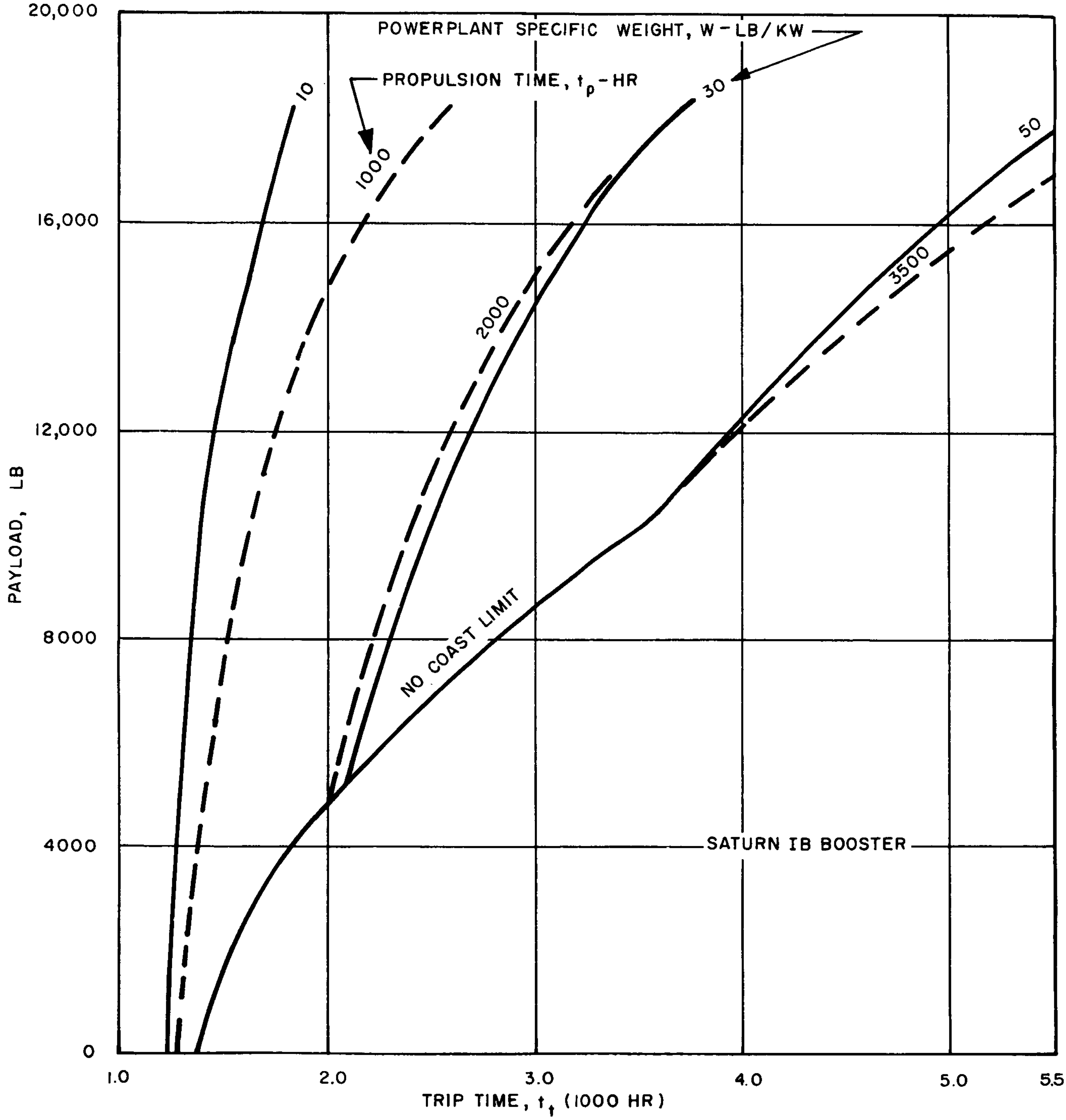


Figure 6.2-5. Asteroid Fly-By Performance

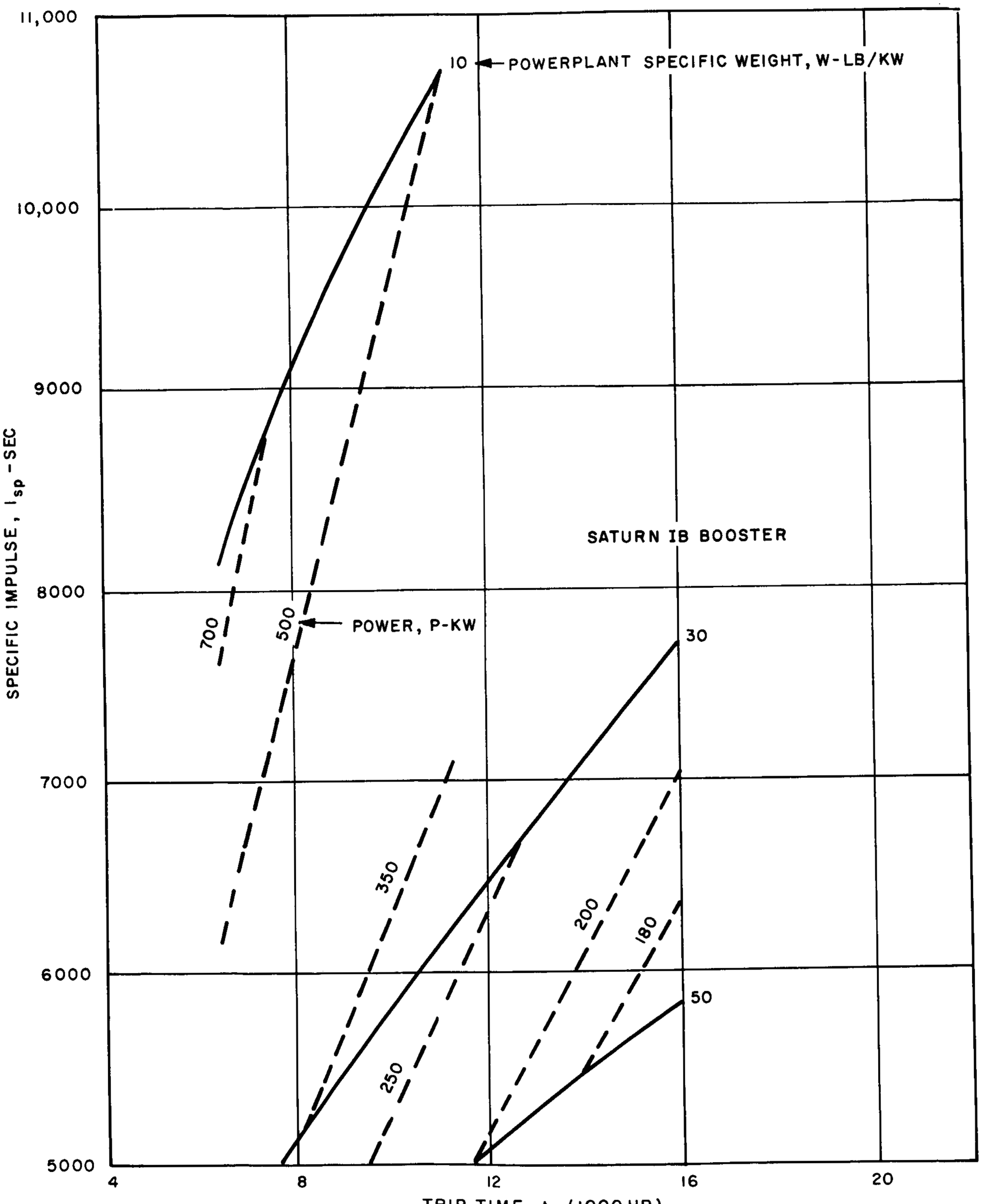


Figure 6.2-6. Asteroid Fly-By Requirements

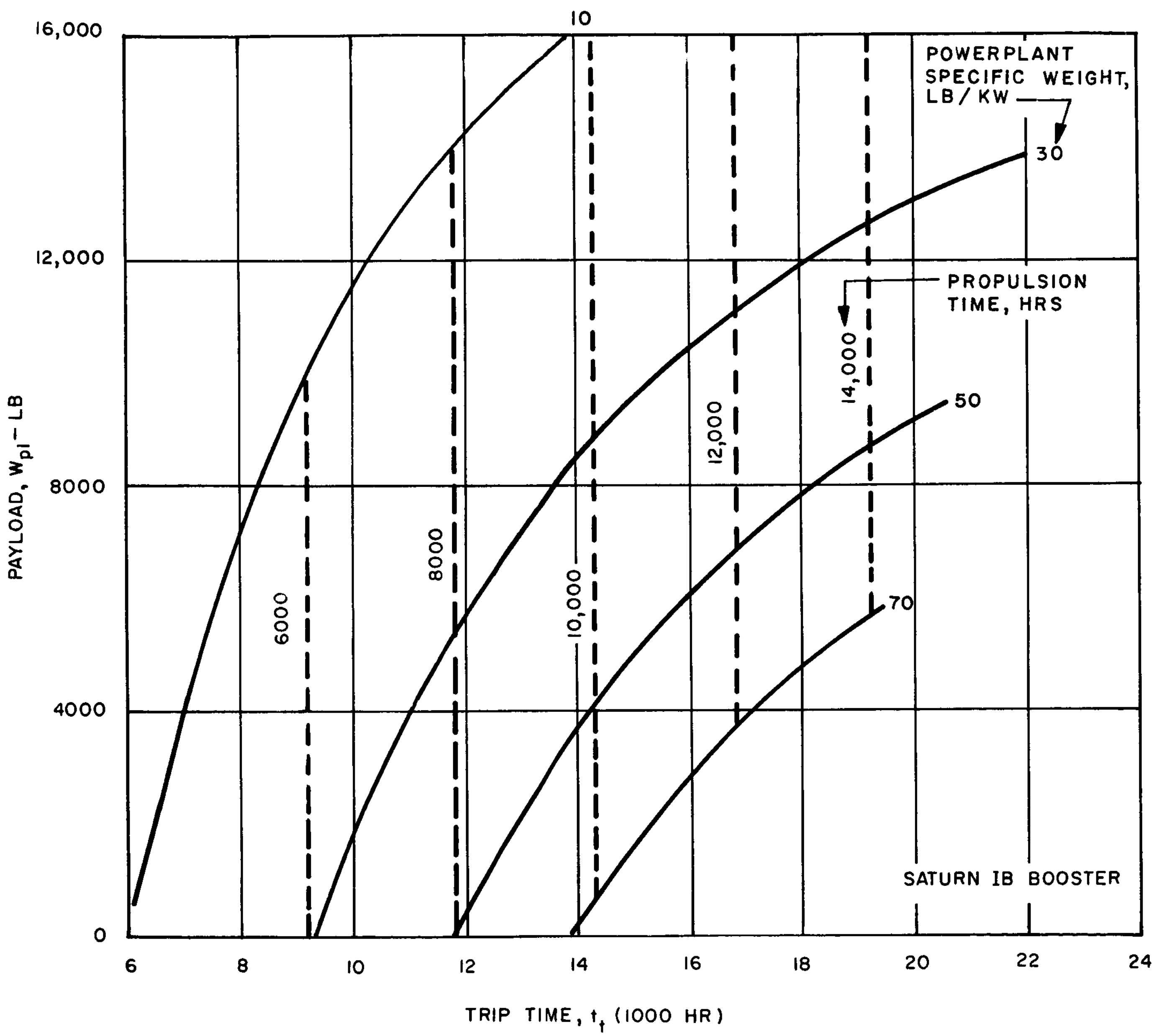


Figure 6.2-7. Jupiter Fly-By Performance

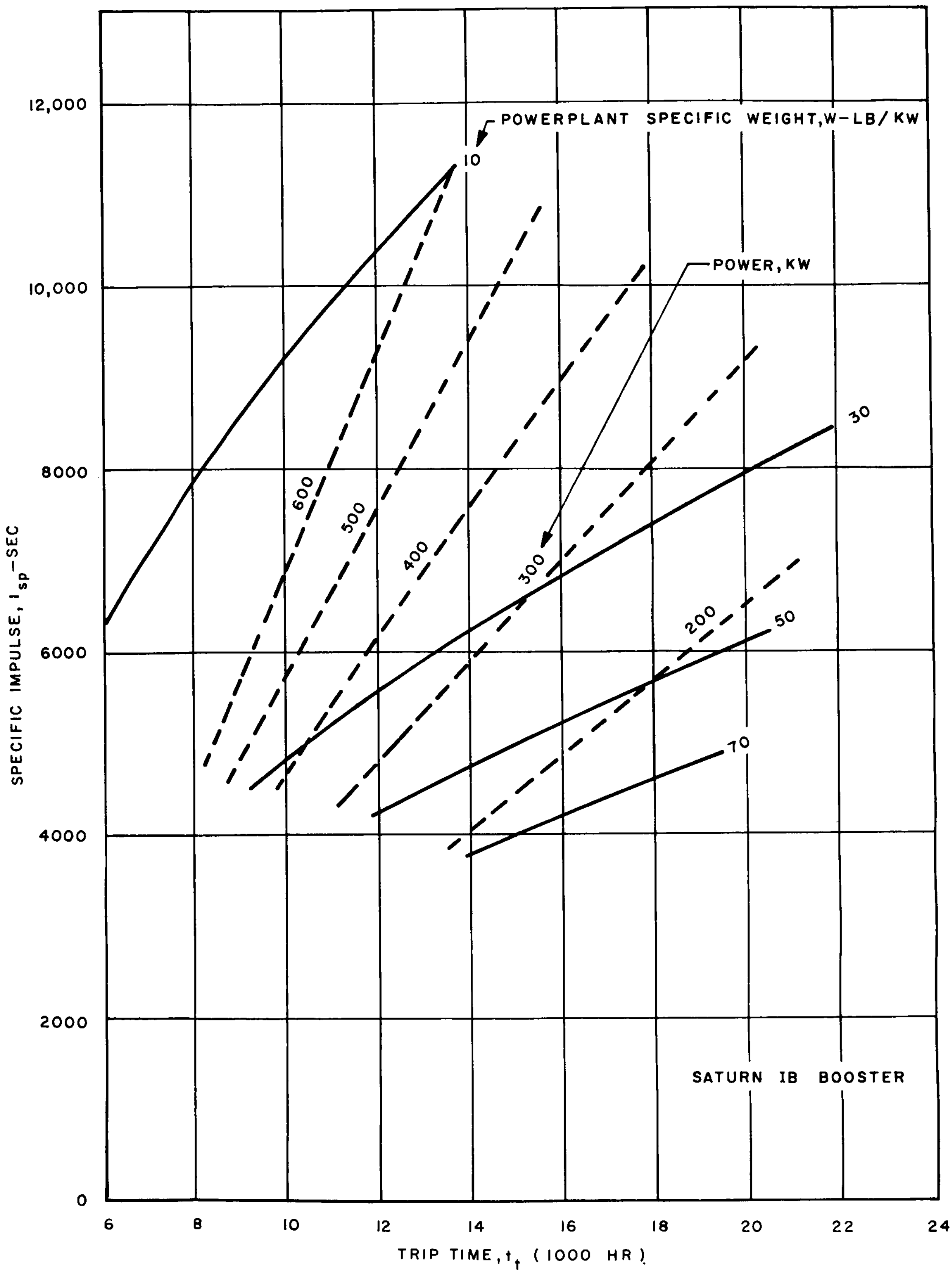


Figure 6.2-8. Jupiter Fly-By Requirements

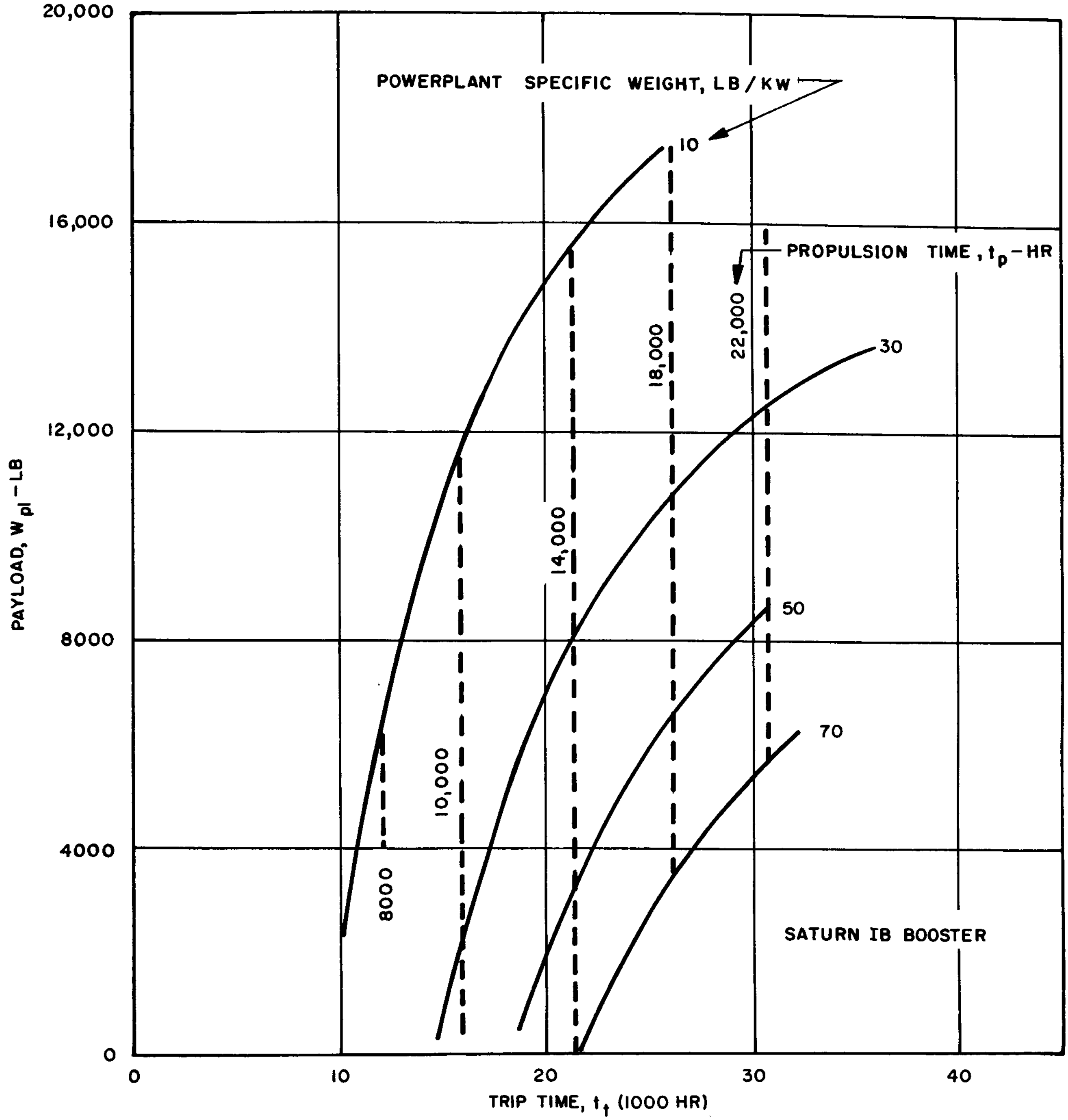


Figure 6.2-9. Saturn Fly-By Performance

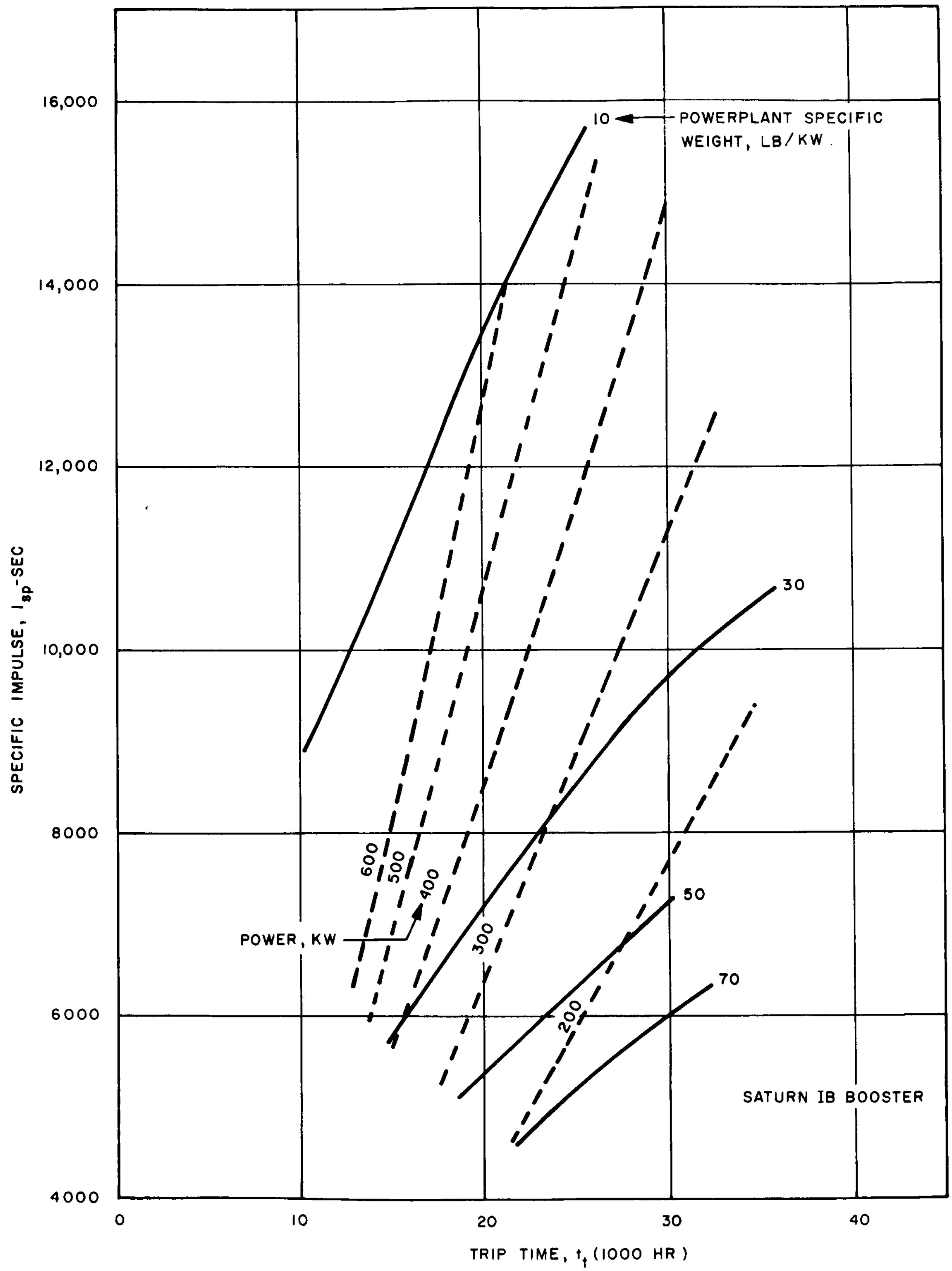


Figure 6.2-10. Saturn Fly-By Requirements

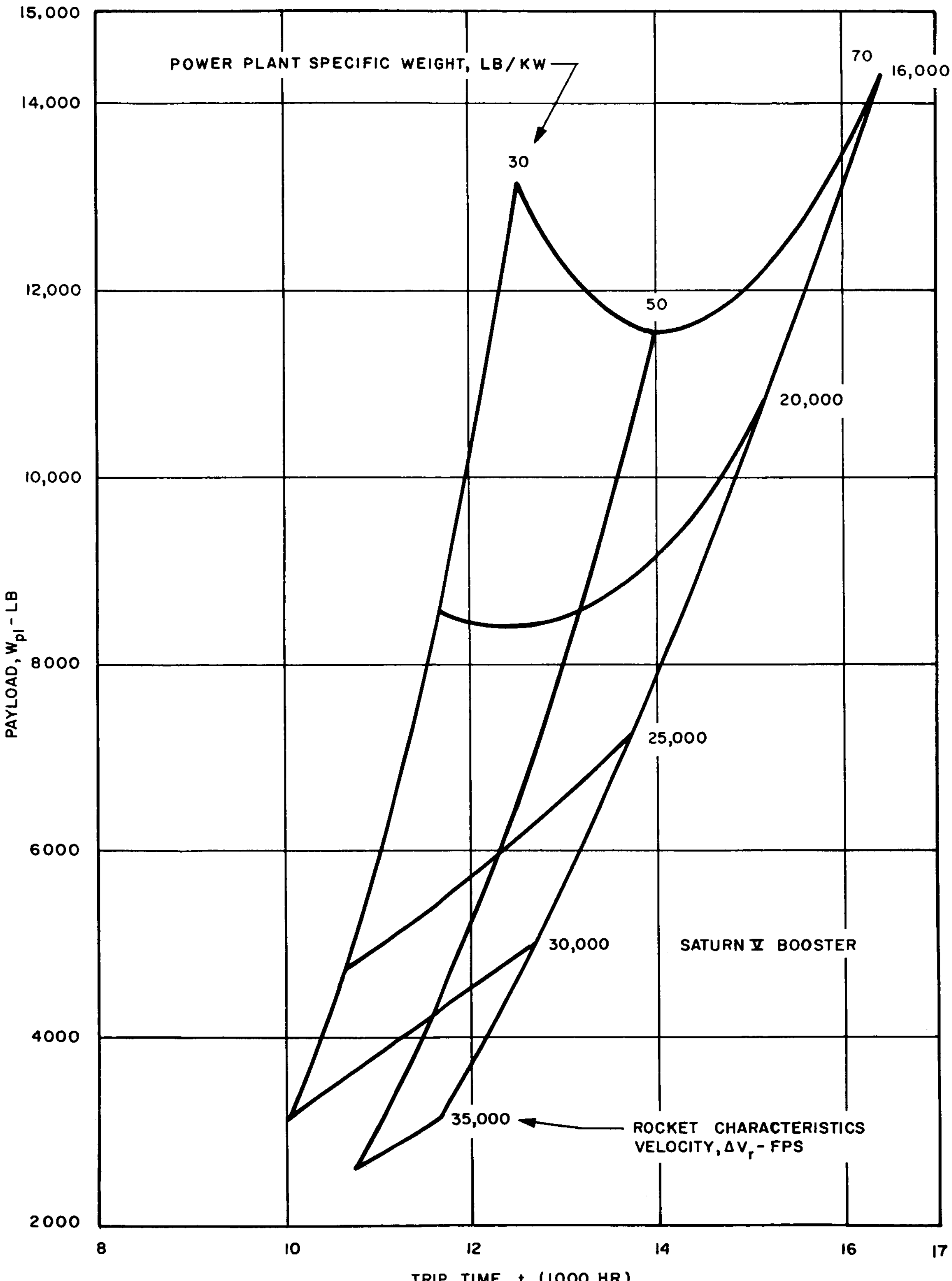


Figure 6.2-11. Saturn Fly-By Performance for 10,000-Hour Propulsion

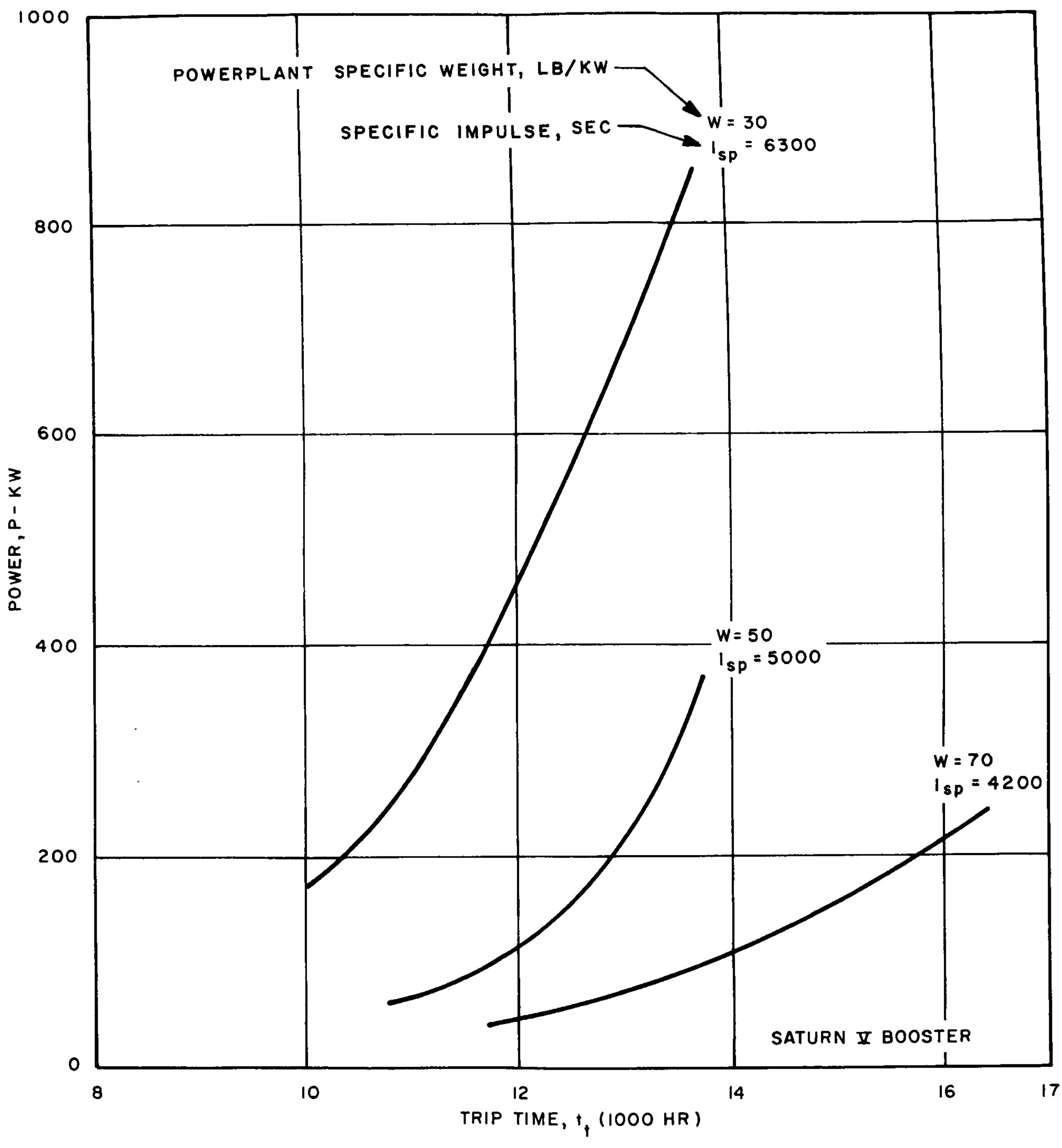


Figure 6.2-12. Saturn Fly-By Requirements for 10,000-Hour Propulsion

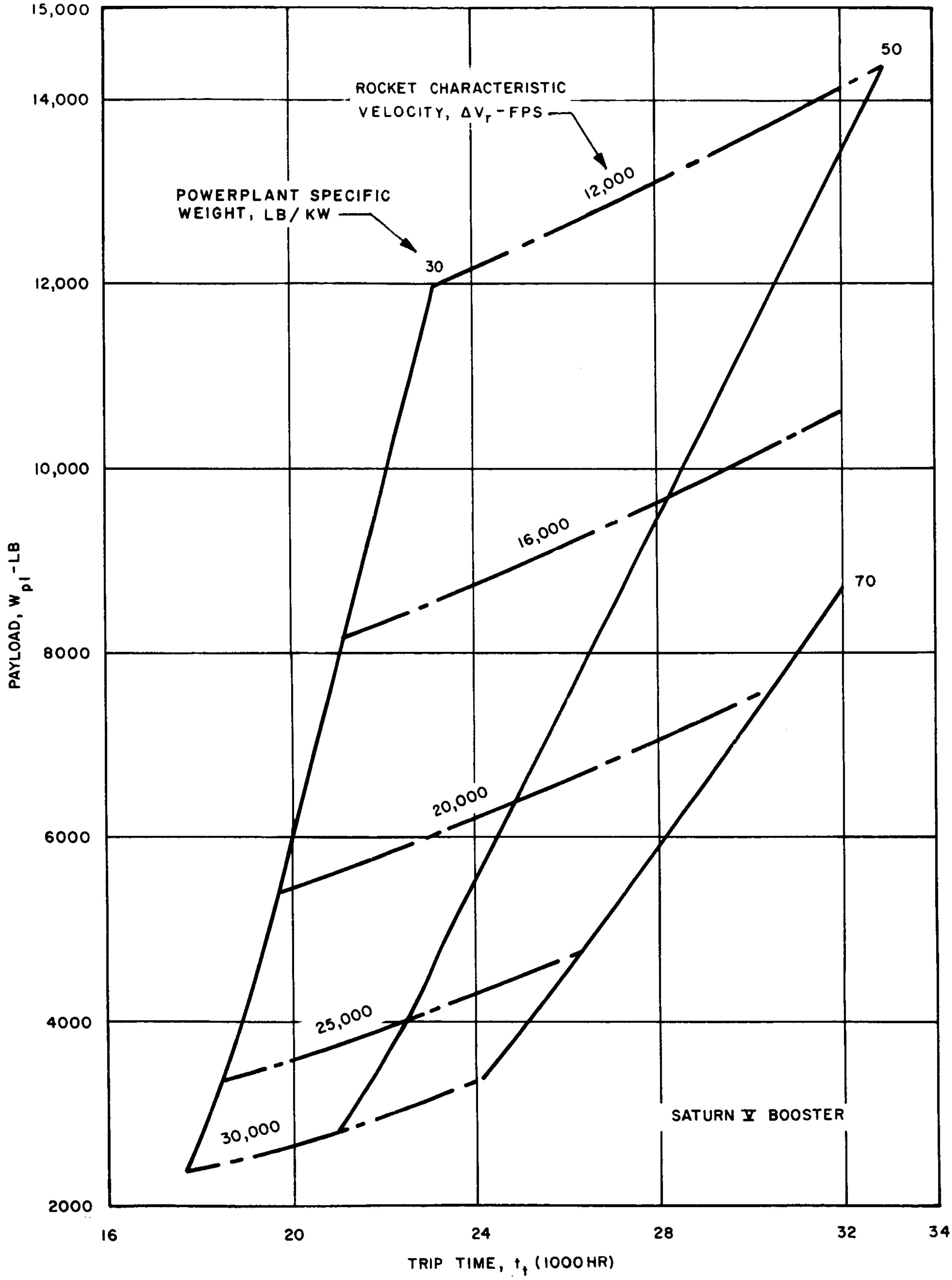


Figure 6.2-13. Uranus Fly-By Performance for 10,000-Hour Propulsion

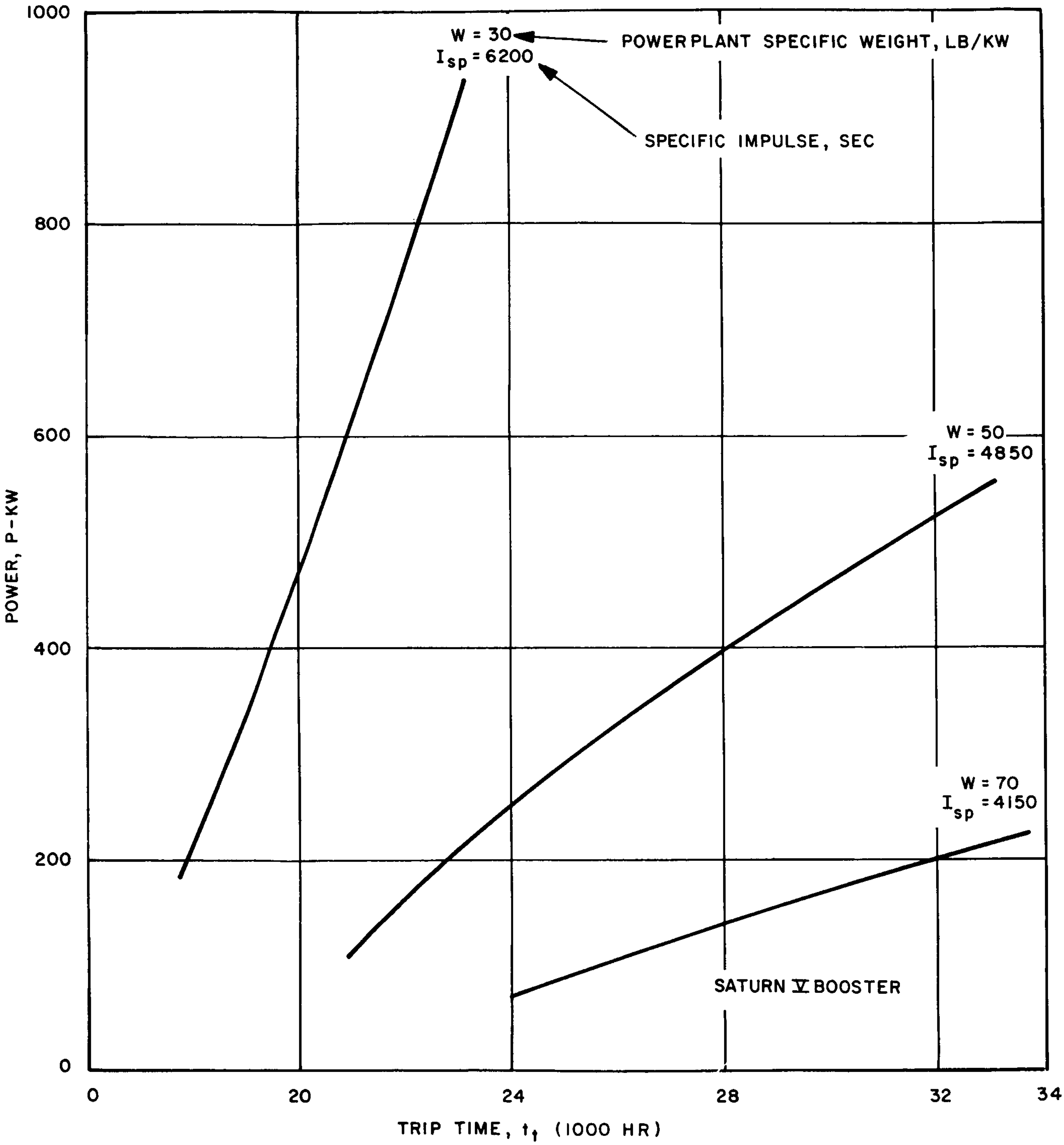


Figure 6.2-14. Uranus Fly-By Requirements for 10,000-Hour Propulsion

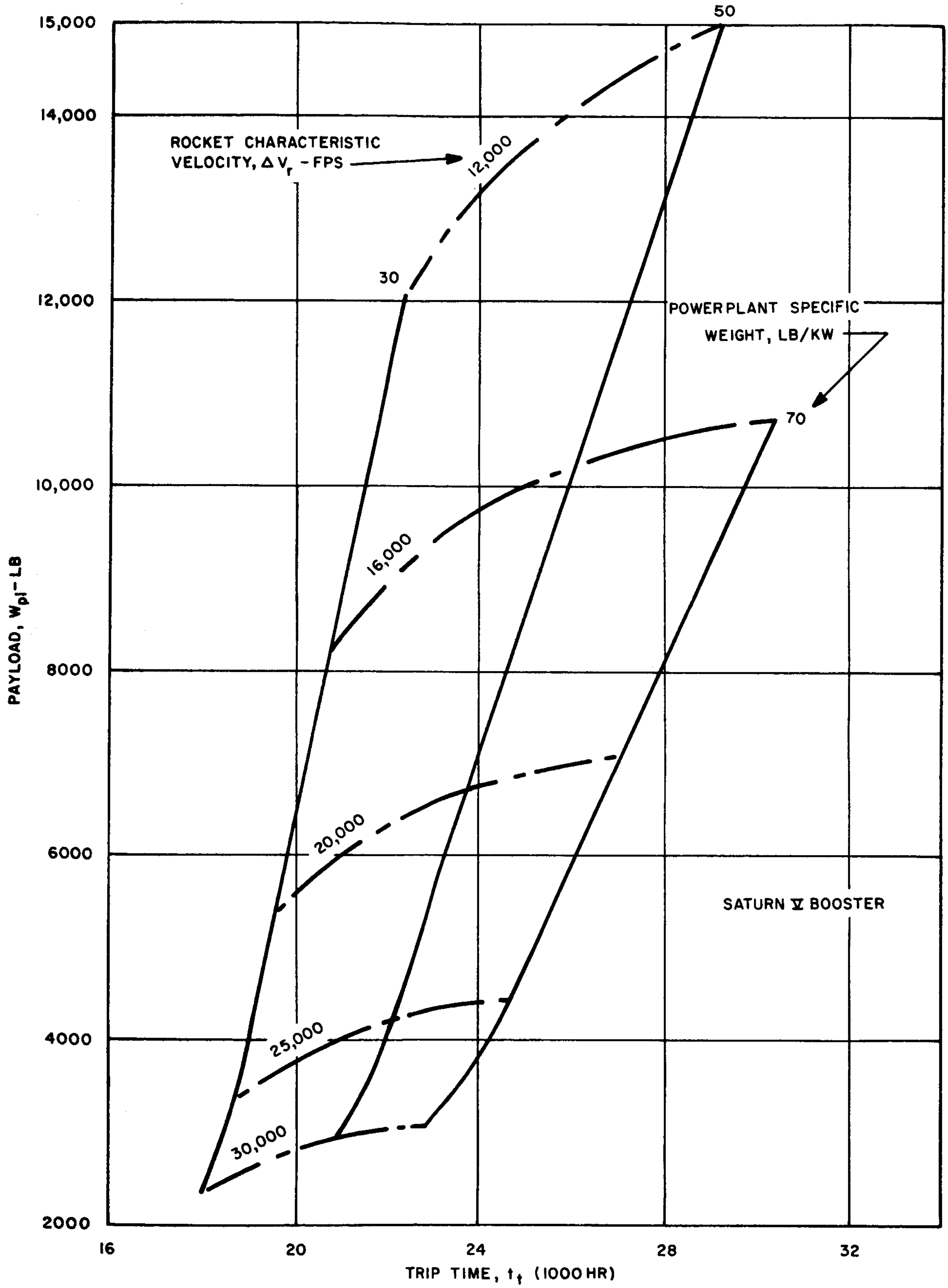


Figure 6.2-15. Uranus Fly-By Performance for 15,000-Hour Propulsion

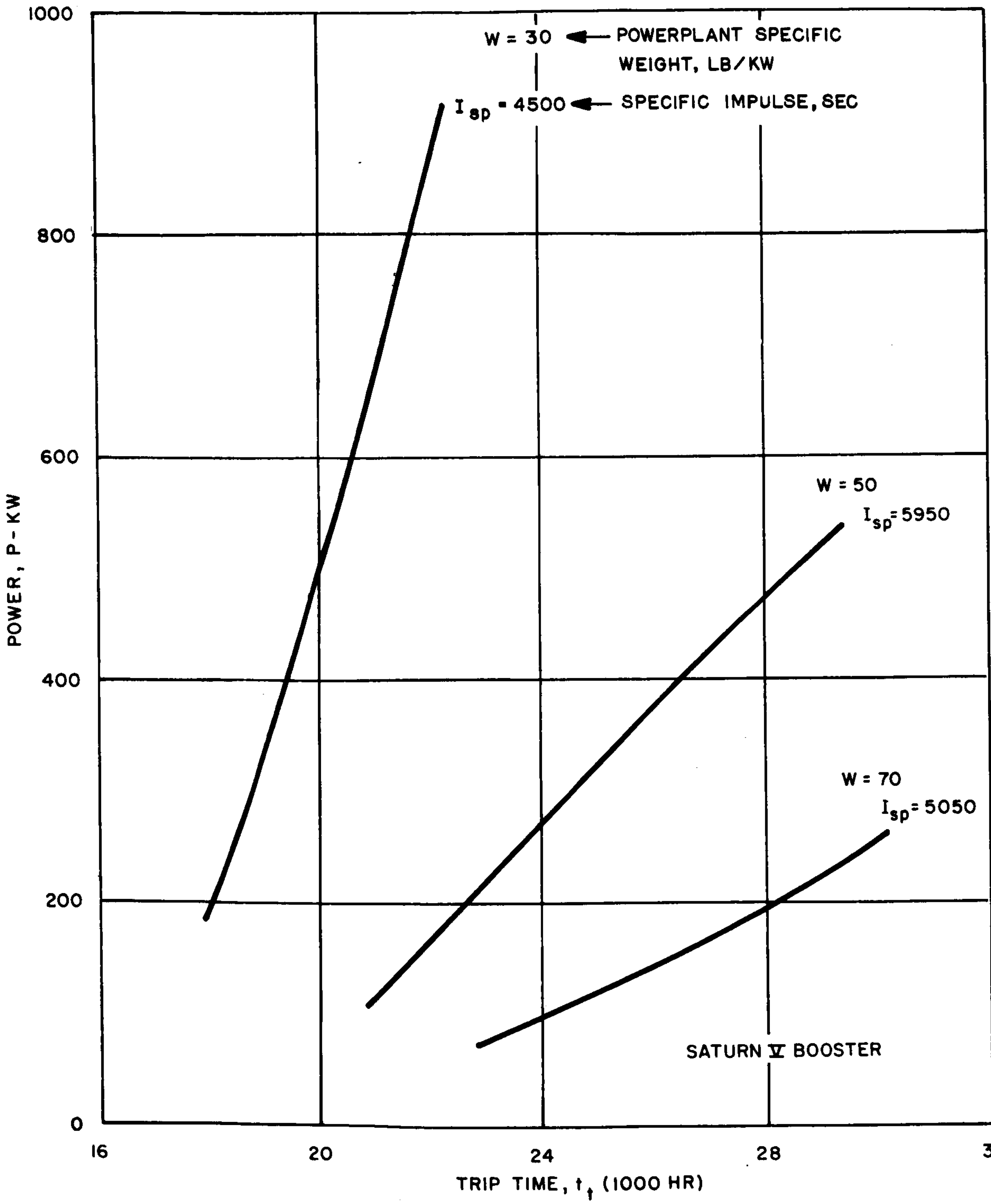


Figure 6.2-16. Uranus Fly-By Requirements for 15,000-Hour Propulsion

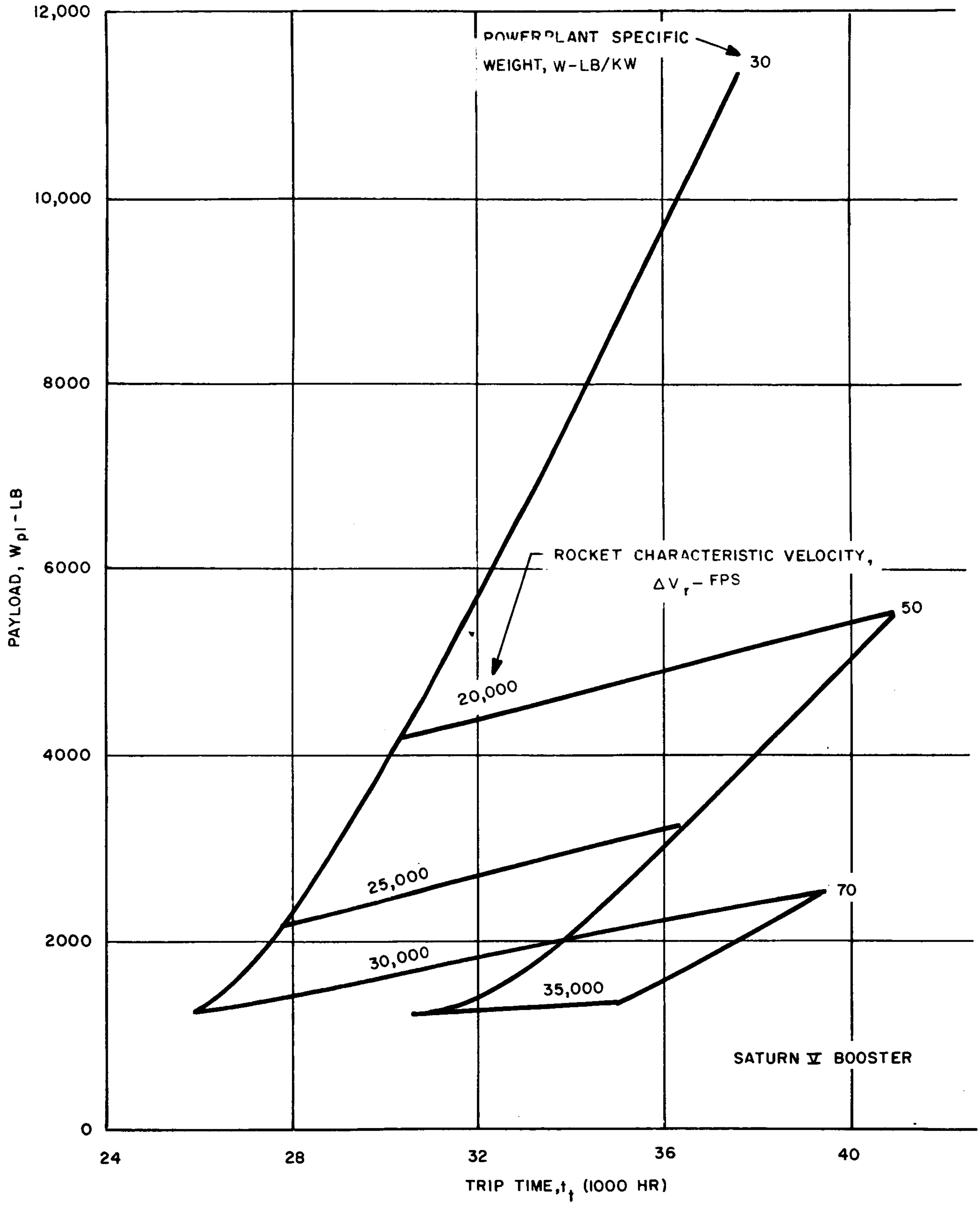


Figure 6.2-17. Neptune and Pluto Fly-By Performance for 10,000 Hour Propulsion

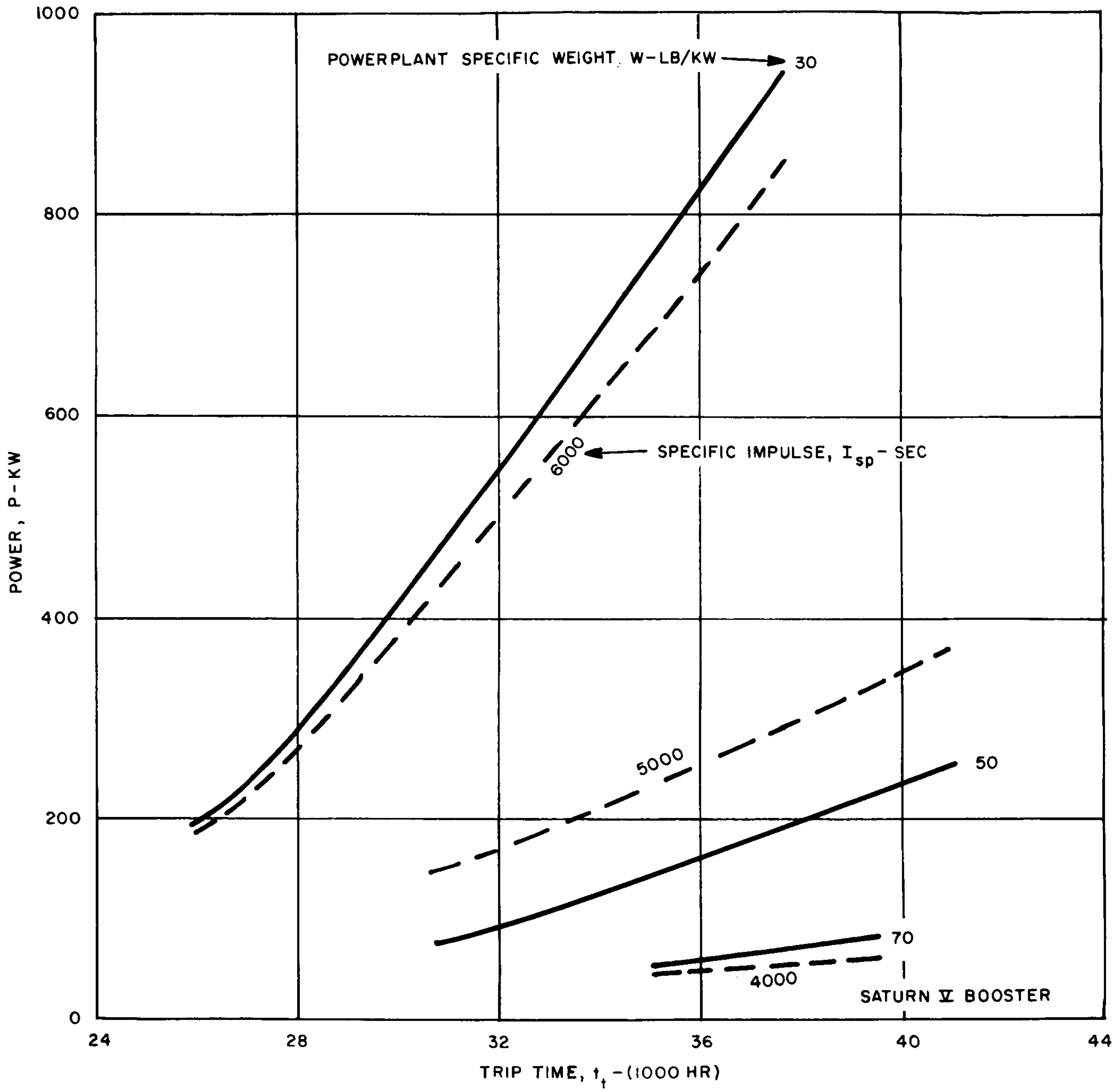


Figure 6.2-18. Neptune and Pluto Fly-By Requirements for 10,000-Hour Propulsion

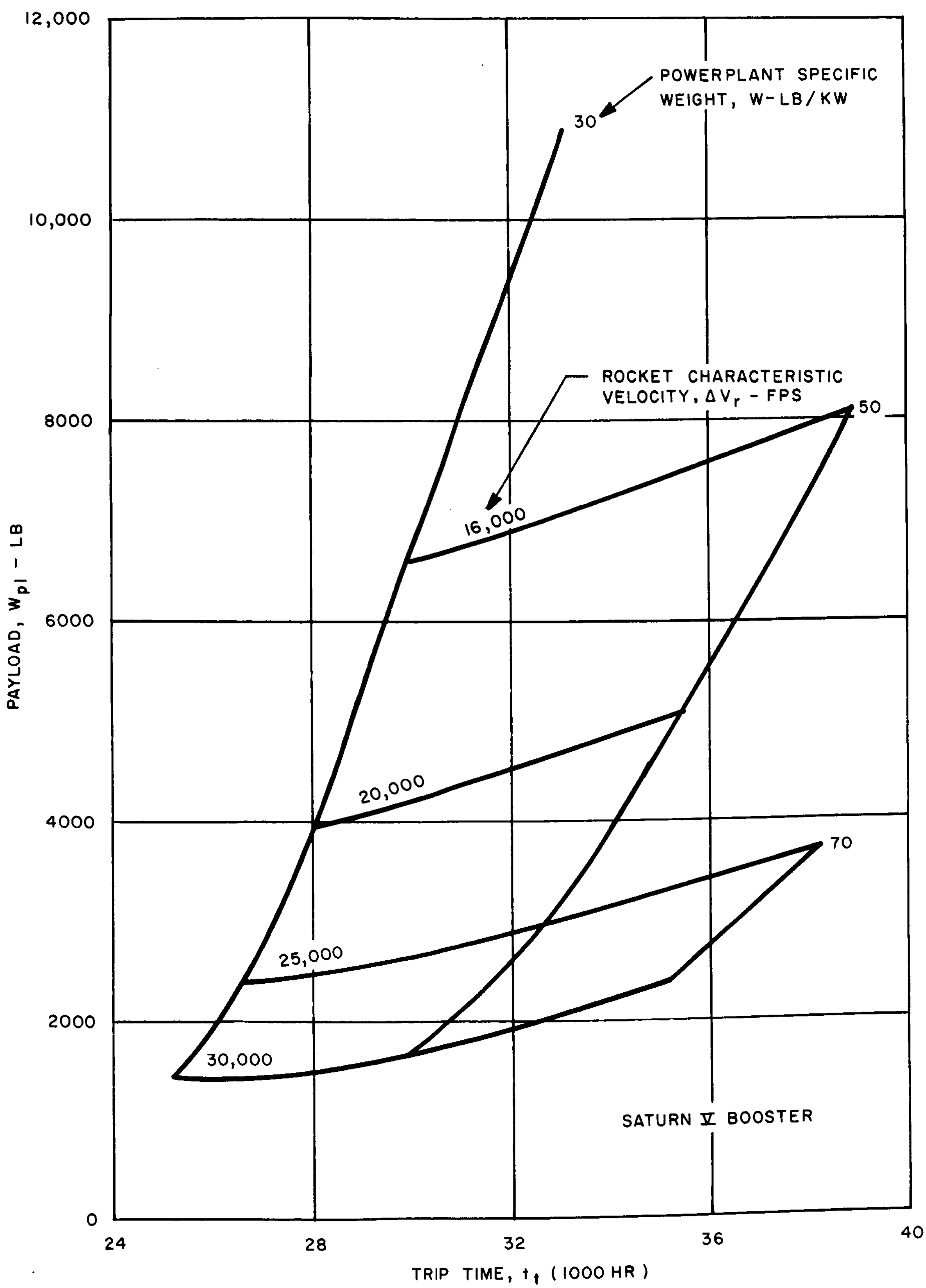


Figure 6.2-19. Neptune and Pluto Fly-By Performance for 15,000-Hour Propulsion

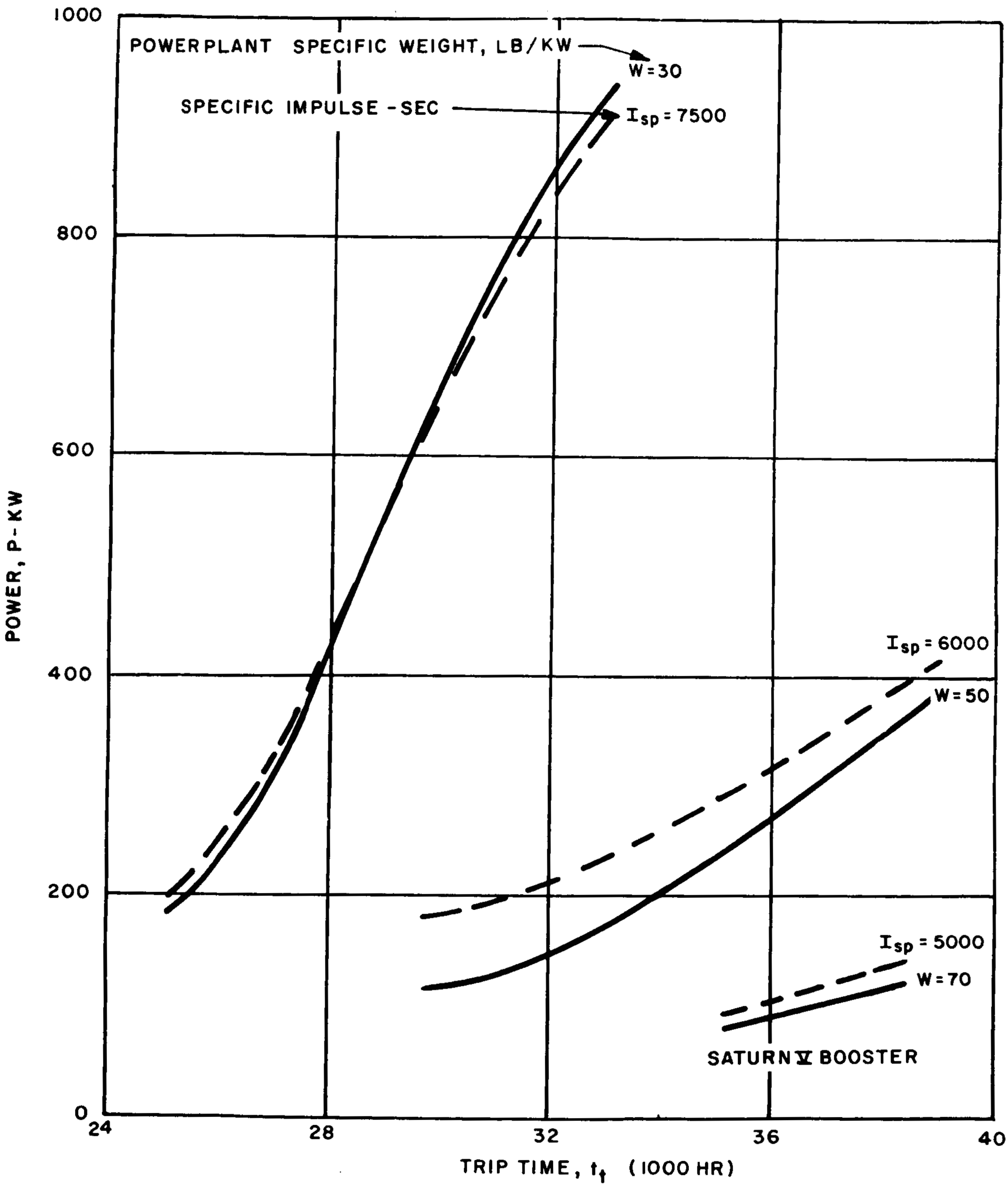


Figure 6.2-20. Neptune and Pluto Fly-By Requirements for 15,000-Hour Propulsion

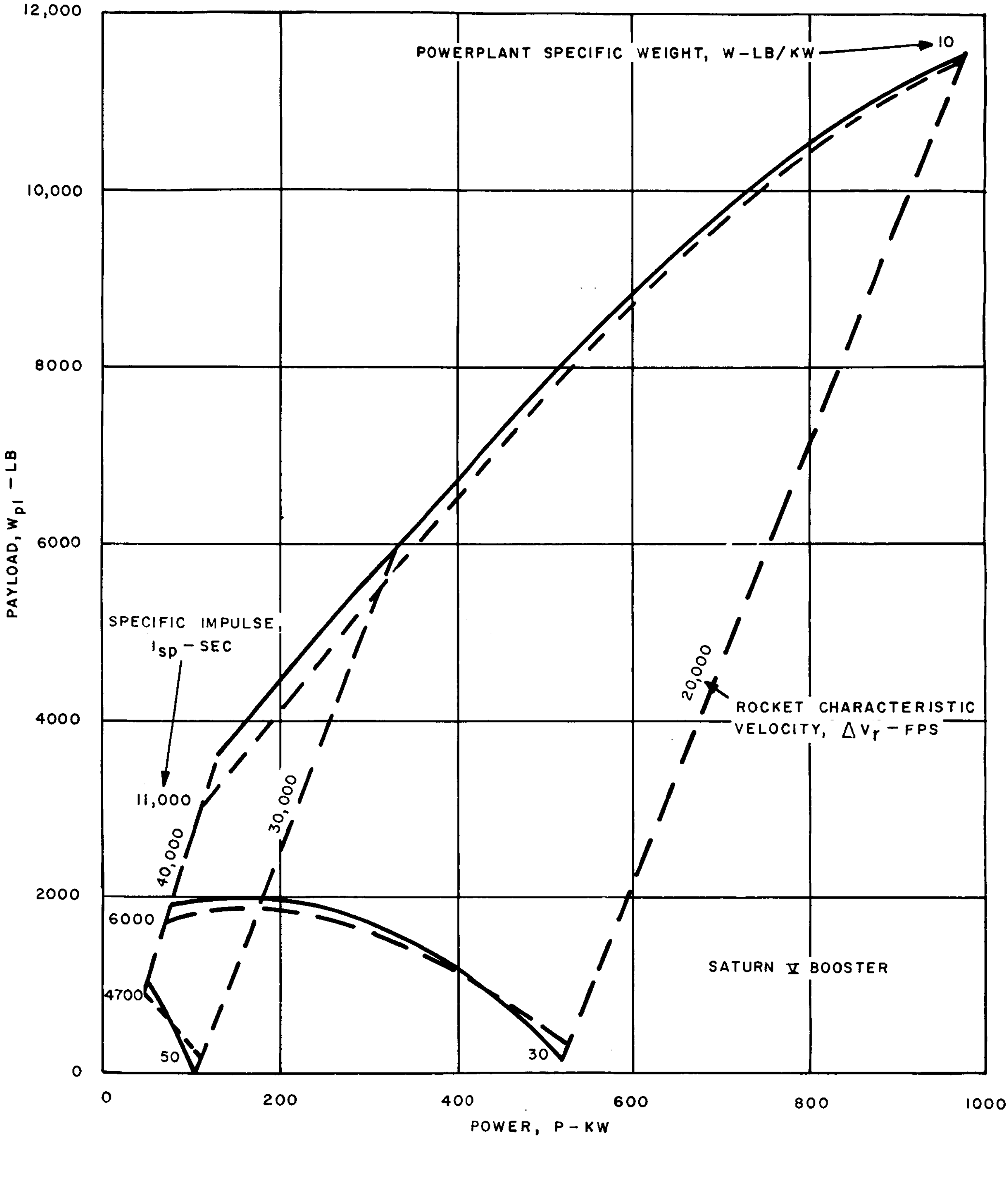


Figure 6.2-21. 35 Degree Out-of-the-Ecliptic Fly-By Performance for 10,000-Hour Propulsion

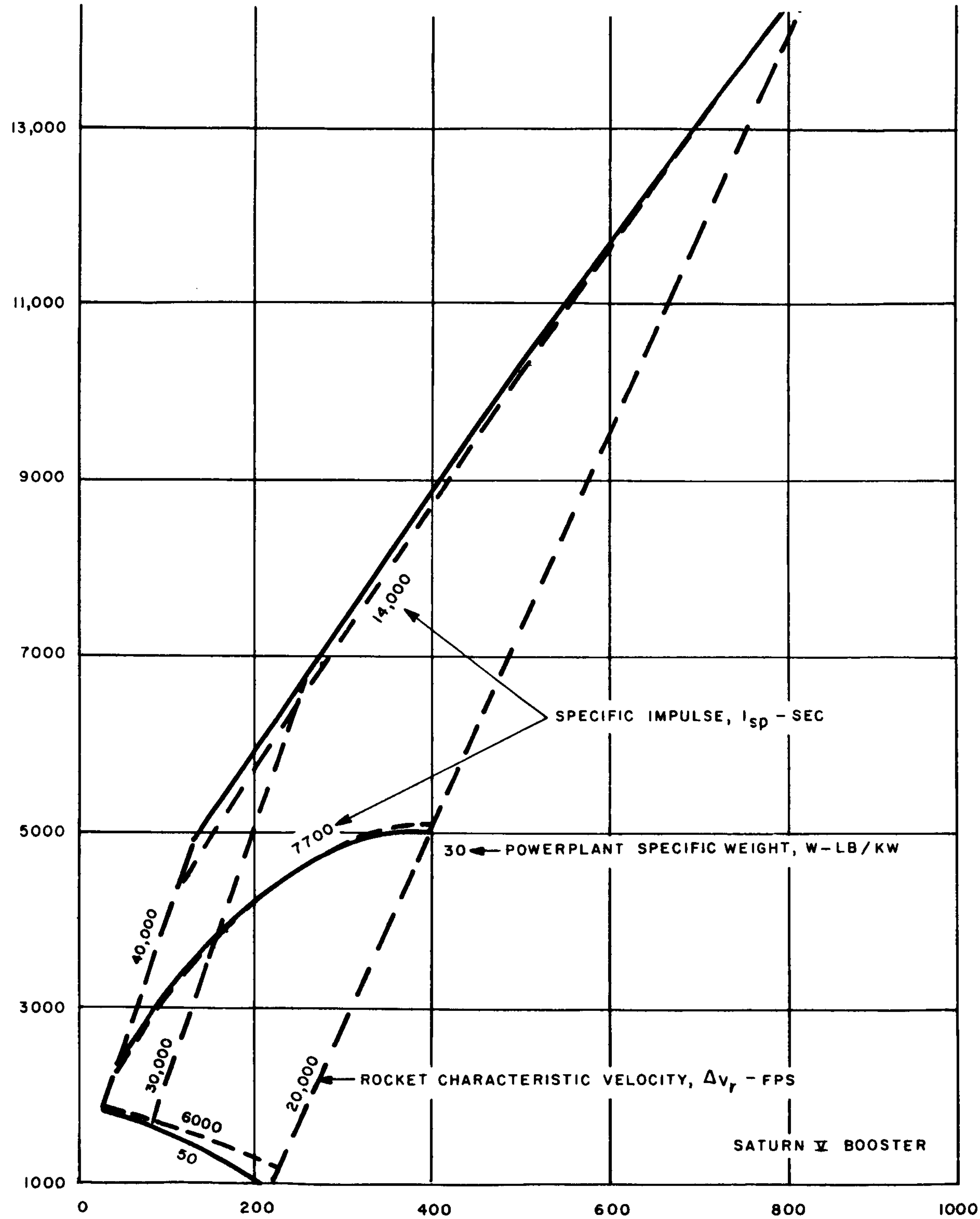


Figure 6.2-22. 35 Degree Out-of-the-Ecliptic Fly-By Performance for 15,000-Hour Propulsion

6.2.2 ORBITER MISSIONS

The performance maps for the planetary orbiter missions are contained in Figures 6.2-23 through 6.2-42. These missions utilize the Saturn V boost vehicle with one to two stages of initial high thrust orbital propulsion. A maximum rocket characteristic velocity of 40,000 fps has, however, been imposed on this phase of the investigation. The gross weight associated with a higher characteristic velocity would not accommodate the NAVIGATOR type neclear-electric powerplant.

Figure 6.2-23 summarizes the performance for the Mercury orbiter mission. Payload has been plotted against trip time with parameters of constant powerplant specific weight, propulsion time, and rocket characteristic velocity. Note that payload is essentially dependent only on the level of rocket characteristic velocity and that trip time is mainly dependent upon the powerplant specific weight. The no coast limit represents a severe limitation on operation at 70 pounds per kw which has therefore been omitted. Figure 6.2-24 contains the companion curve showing the variation in optimum power requirements. Optimum specific impulse is again maintained constant at 3000 seconds.

Similar data is contained in Figures 6.2-25 through 6.2-28 for the Venus and Mars orbiter missions. Here, the 10 pound per kw line has been omitted due to excessive power requirements. Note that the propulsion time requirements range from 1000 to 5000 hours for all of the minor planet orbiters.

Figure 6.2-29 summarizes the performance of the Jupiter I orbiter mission using the same format of Figure 6.2-23. The mission characteristics differ from the previous curve, however, in that the payload varies with specific weight as well as the rocket characteristic velocity. Note that propulsion requirements range from about 3000 to 9000 hours. Figure 6.2-30 contains the associated optimum specific impulse requirements as a function of trip time and powerplant specific weight. Constant power requirements are superimposed on these data.

Figures 6.2-31 and 6.2-32 contain similar data on the Jupiter II orbiter mission. Propulsion time requirements for this mission range from 5000 to 15,000 hours. The power requirements are, however, about the same.

Figures 6.2-33 through 6.2-42 complete the performance maps for the remaining orbiter missions - Saturn I, Saturn II, Uranus, Neptune, and Pluto. Although useful payload is generally possible at 10 pounds per kw within 15,000 hours of propulsion time, the higher powerplant specific weights will involve propulsion time requirements as high as 30,000 hours. It is significant to note that the coasting periods for these missions range from 15,000 hours on the Saturn I mission to several times that value for the more distant missions and that the coasting period represents the elapsed time between launch and start-up of the nuclear-electric powerplant.

6.2.3 PERFORMANCE SUMMARY

Figures 6.2-43 through 6.2-54 contain a series of summary curves for the NAVIGATOR fly-by and orbiter missions. These data are the same data shown in Figures 6.2-1 through 6.2-42. They are rearranged, however, to illustrate the effects of mission difficulty on payload capabilities and on propulsion and trip time requirements. Figure 6.2-43 summarizes all fly-by performance with a powerplant specific weight of 10 pounds per kw. Figures 6.2-44 through 6.2-49 contain similar data for 30, 50, and 70 pounds per kw with the Saturn IB and Saturn V boosters. Figure 6.2-50 summarizes the performance for all of the minor planet orbiters and Figures 6.2-51 through 6.2-54, the performance for the major planet orbiters for 10, 30, 50, and 70 pounds per kw, respectively.

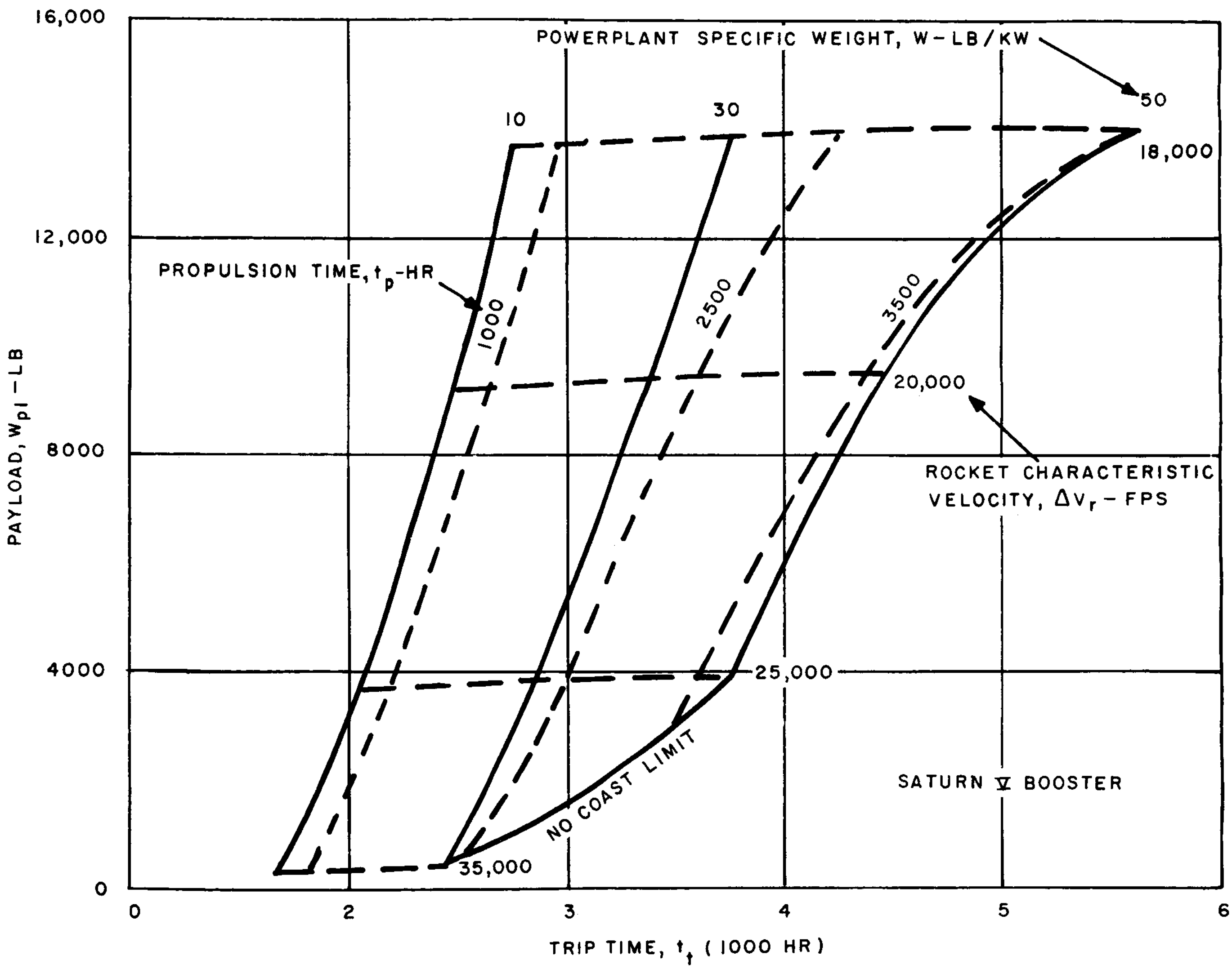


Figure 6.2-23. Mercury Orbiter Performance

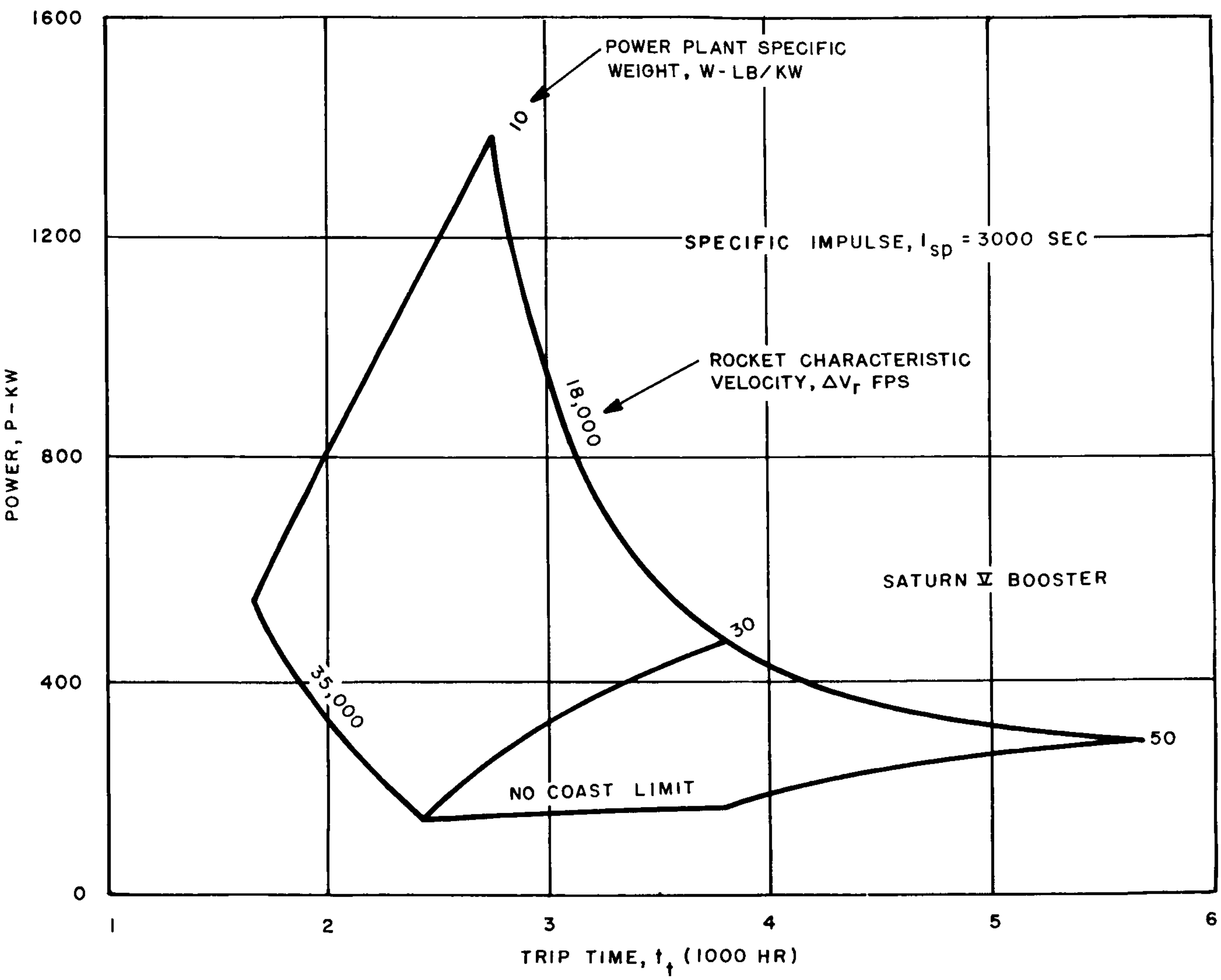


Figure 6.2-24. Mercury Orbiter Requirements

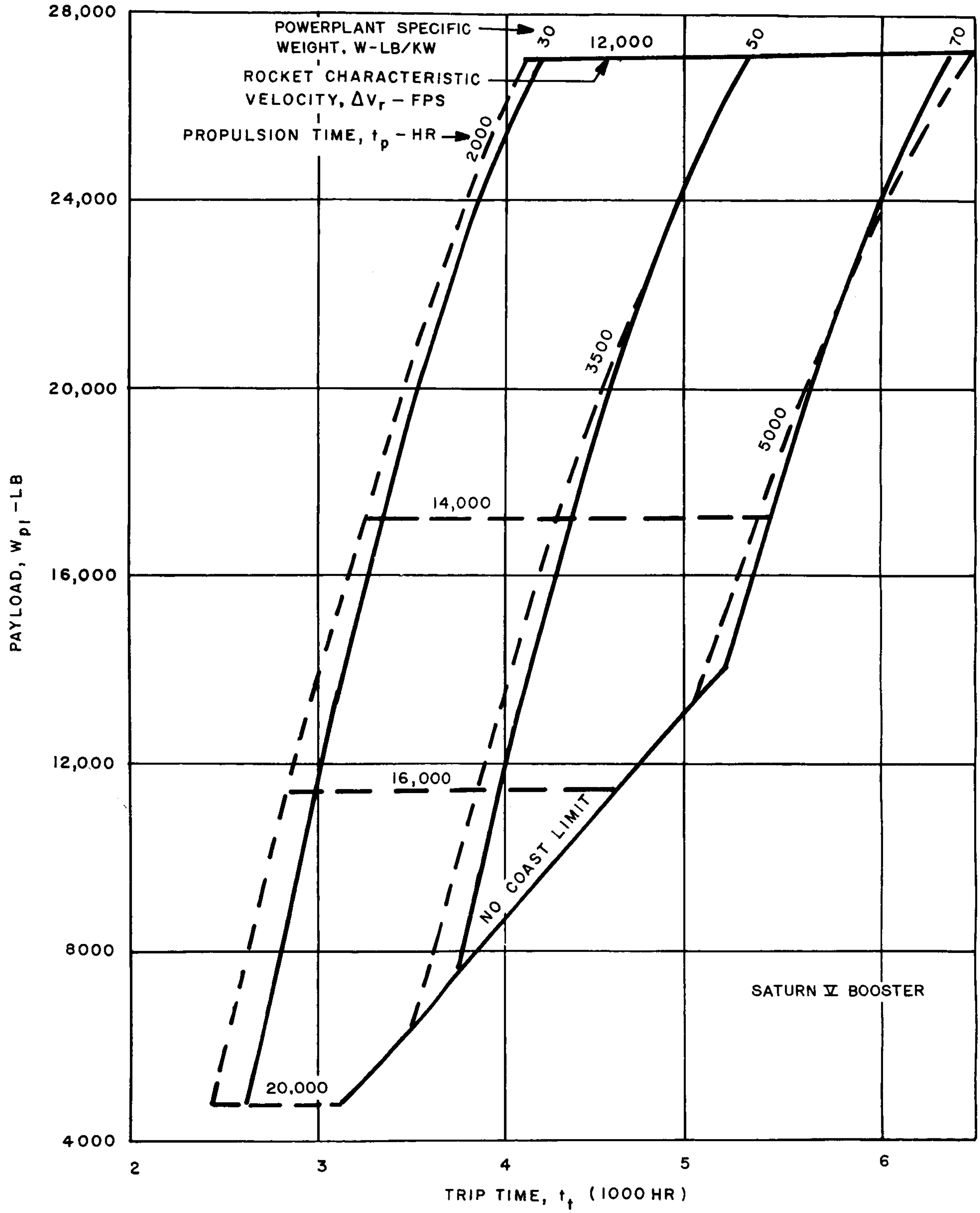


Figure 6.2-25. Venus Orbiter Performance

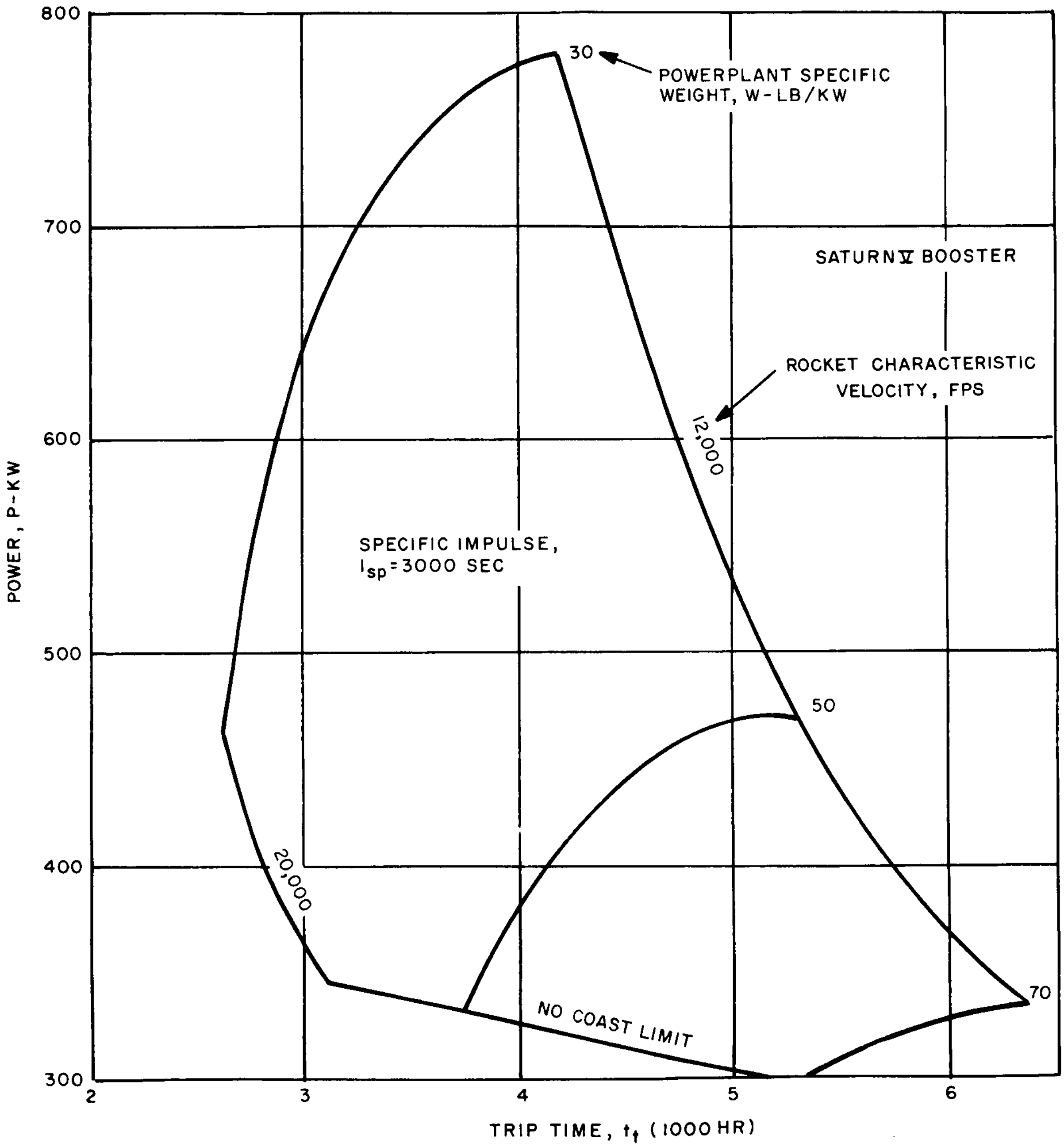


Figure 6.2-26. Venus Orbiter Requirements

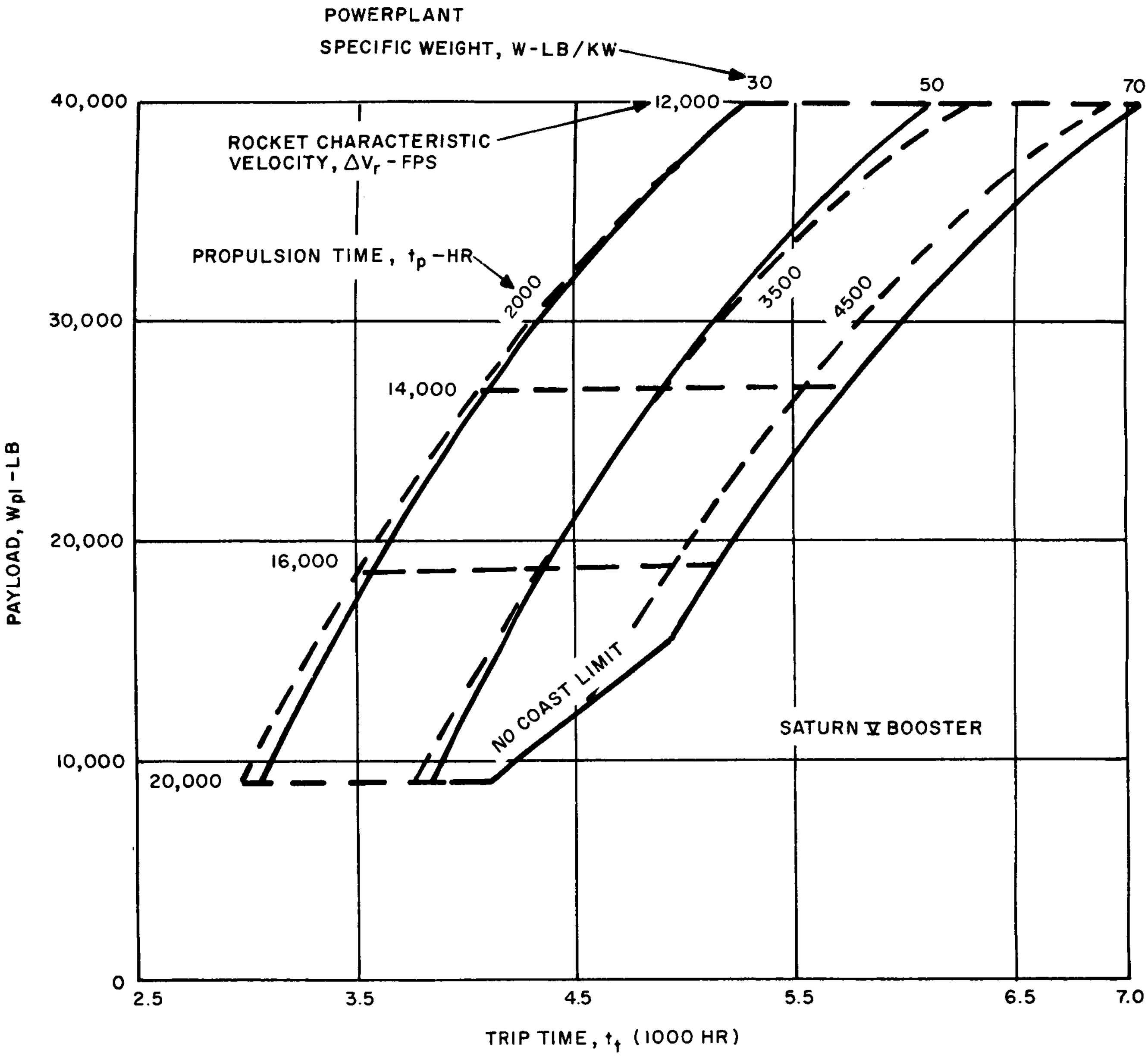


Figure 6.2-27. Mars Orbiter Performance

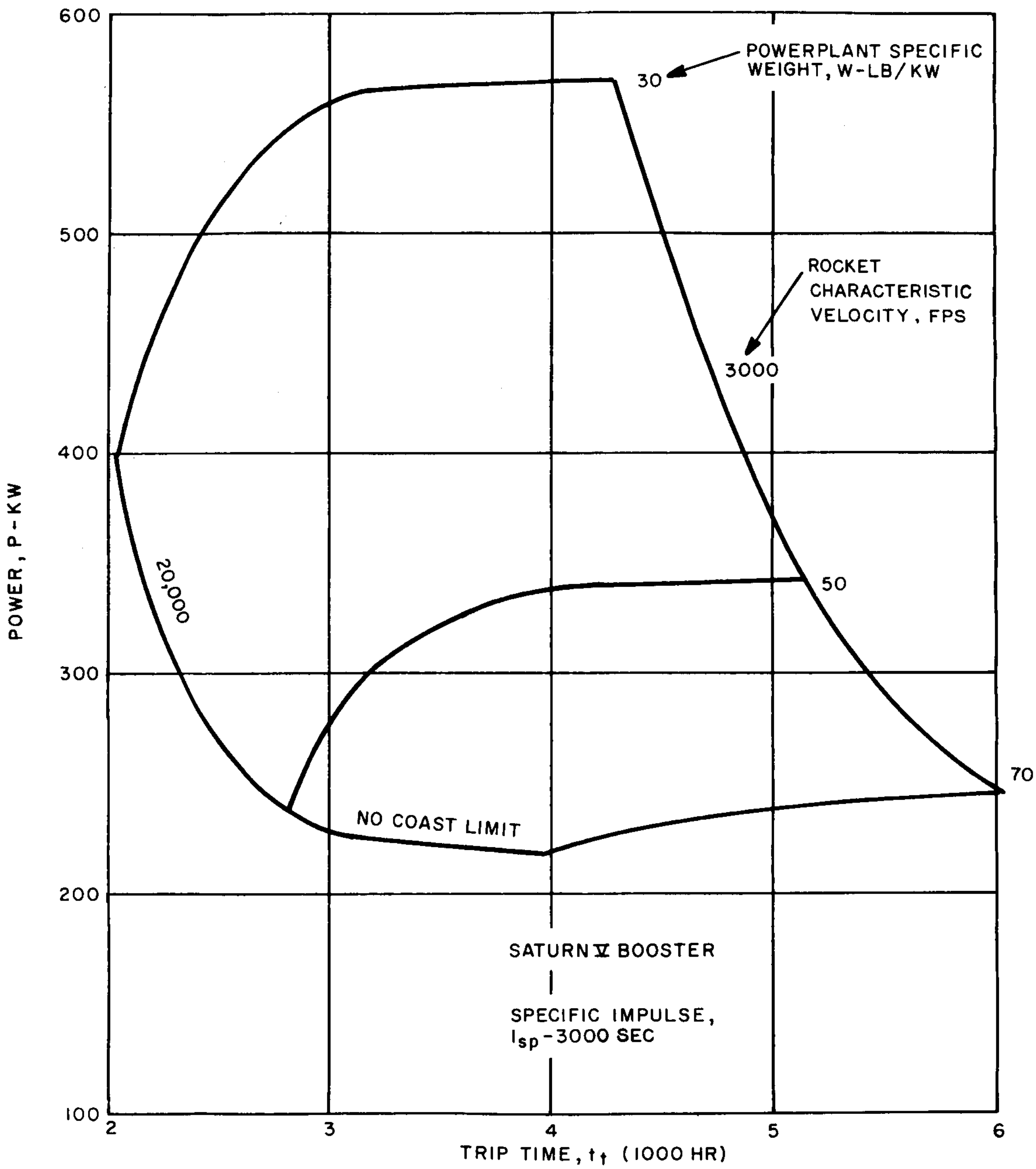


Figure 6.2-28. Mars Orbiter Requirements

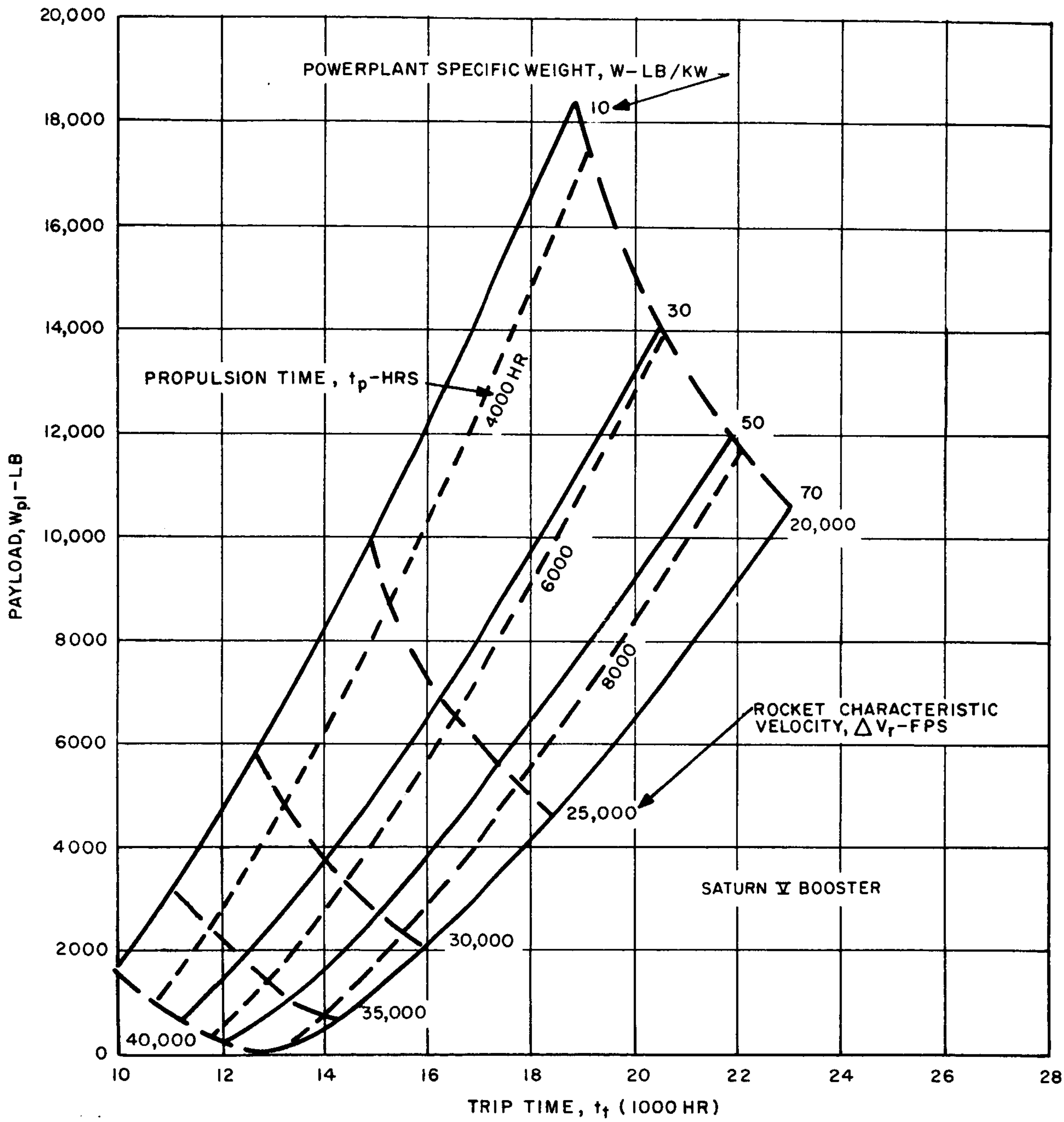


Figure 6.2-29. Jupiter I Orbiter Performance

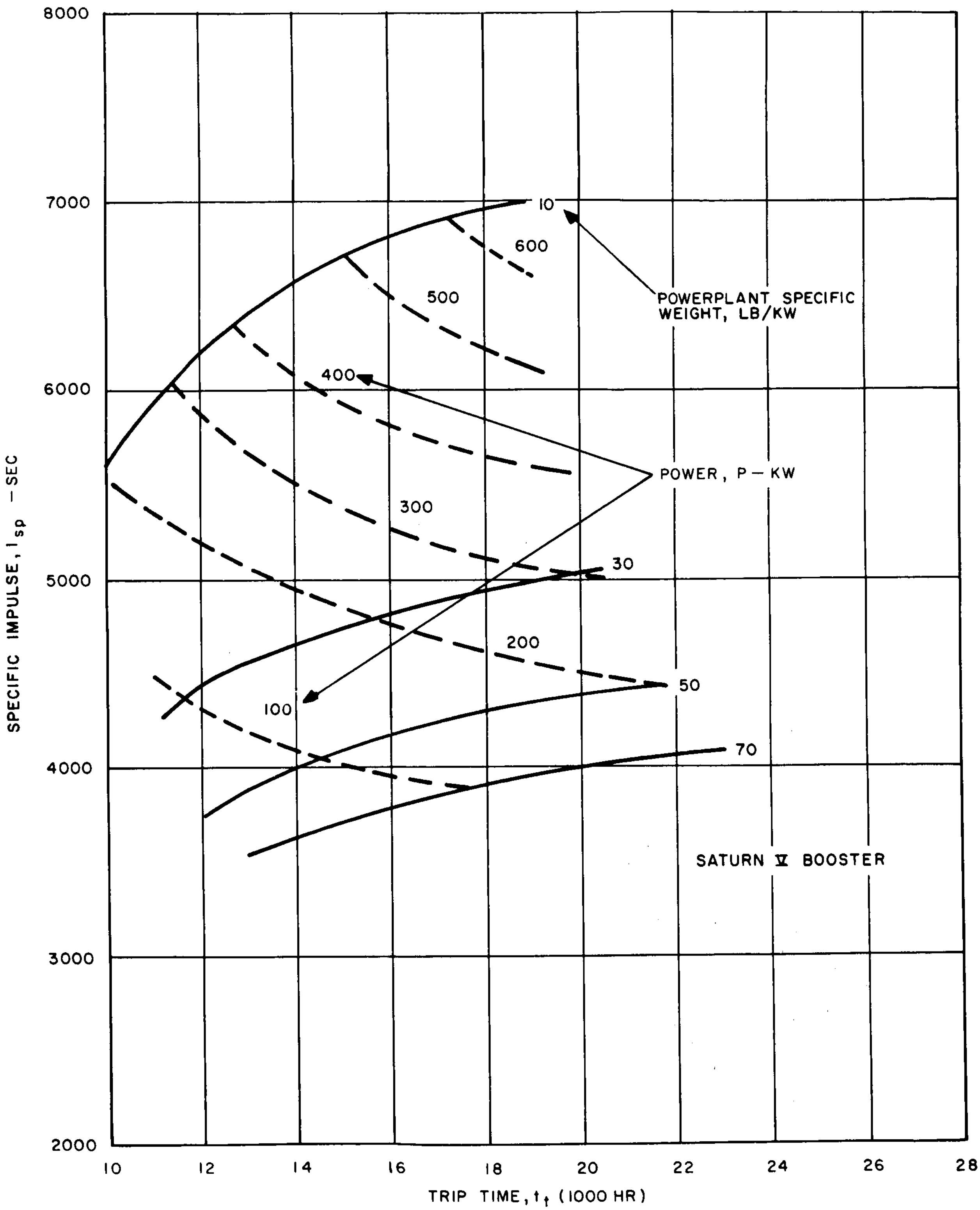


Figure 6.2-30. Jupiter I Orbiter Requirements

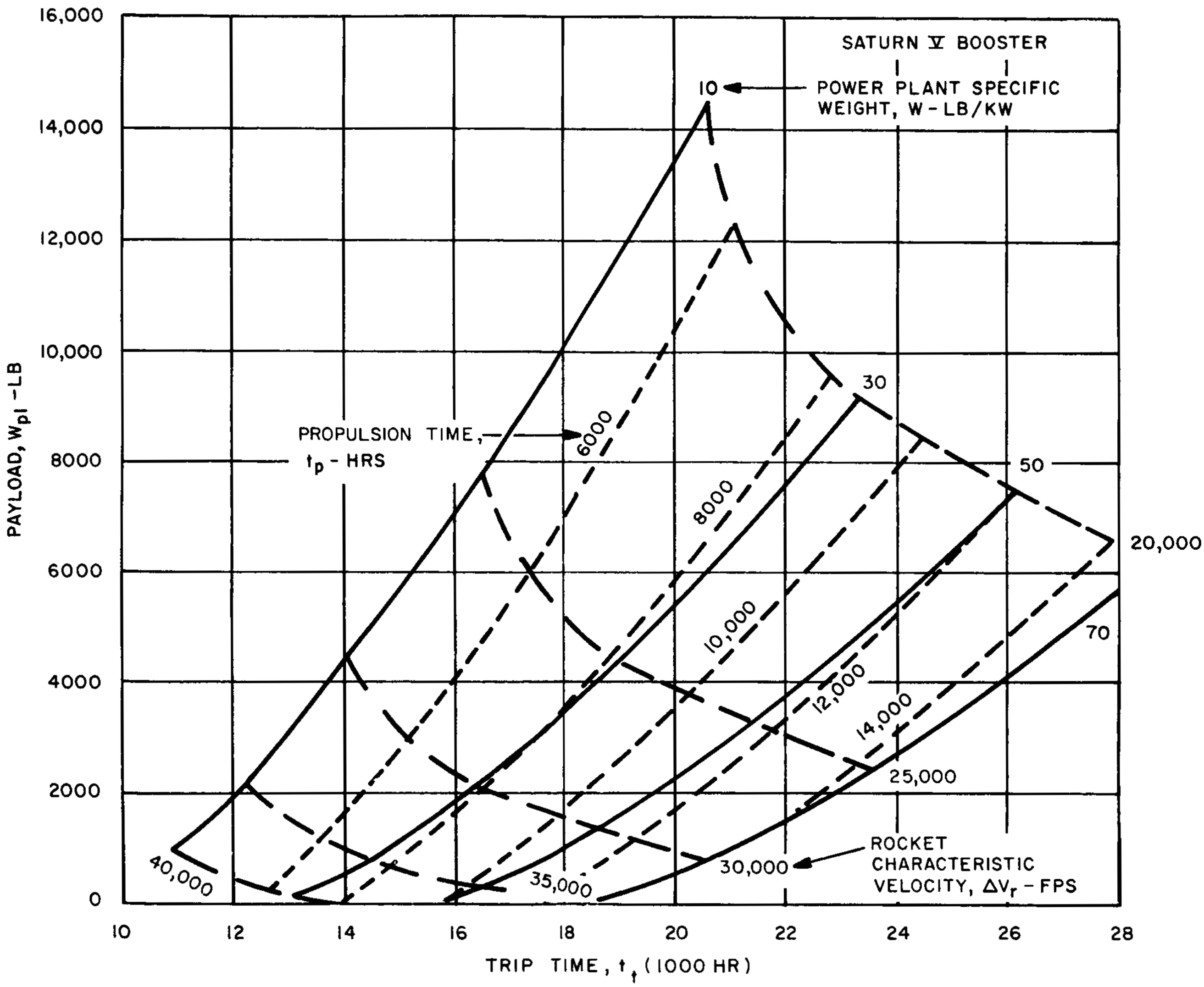


Figure 6.2-31. Jupiter II Orbiter Performance

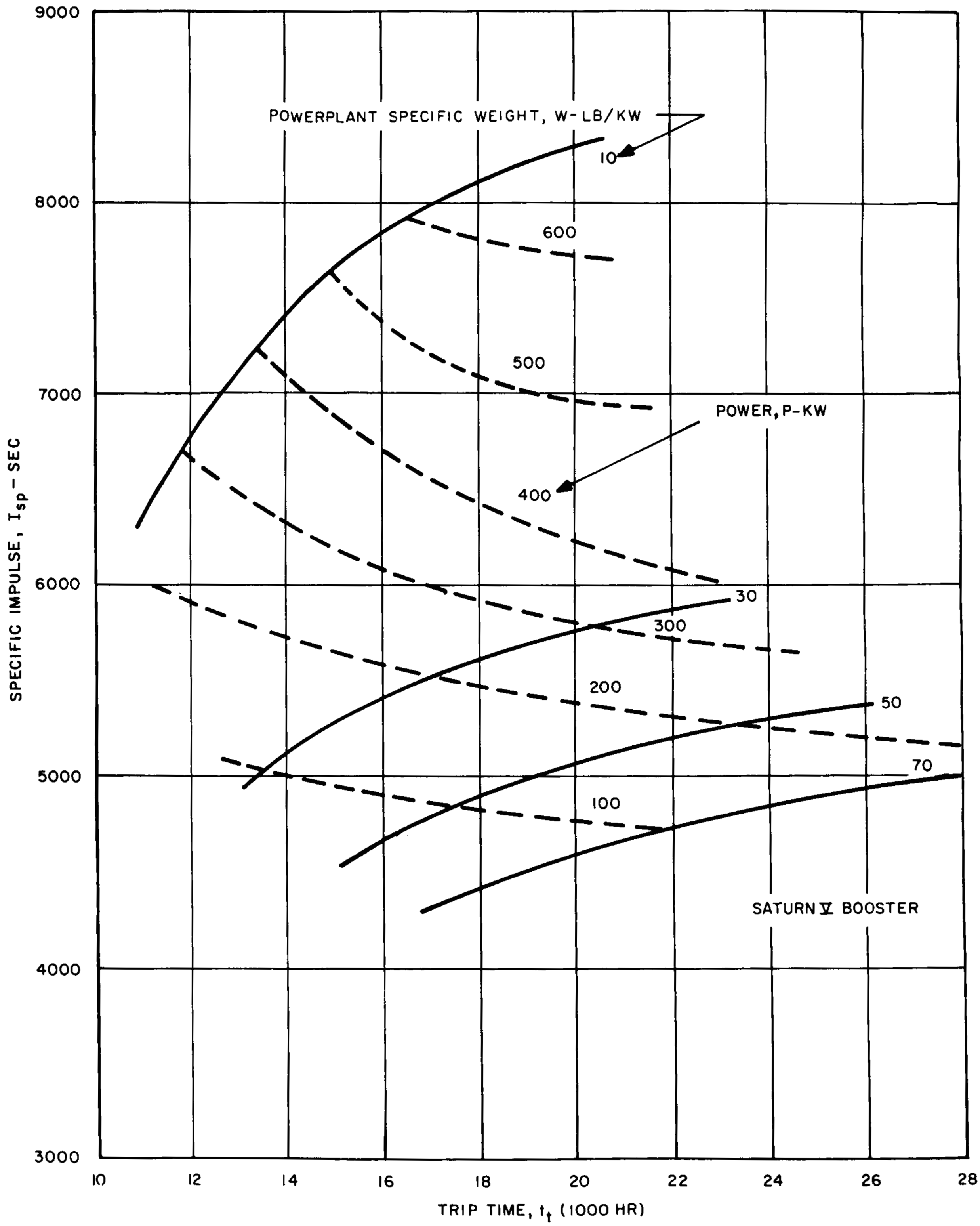


Figure 6.2-32. Jupiter II Orbiter Requirements

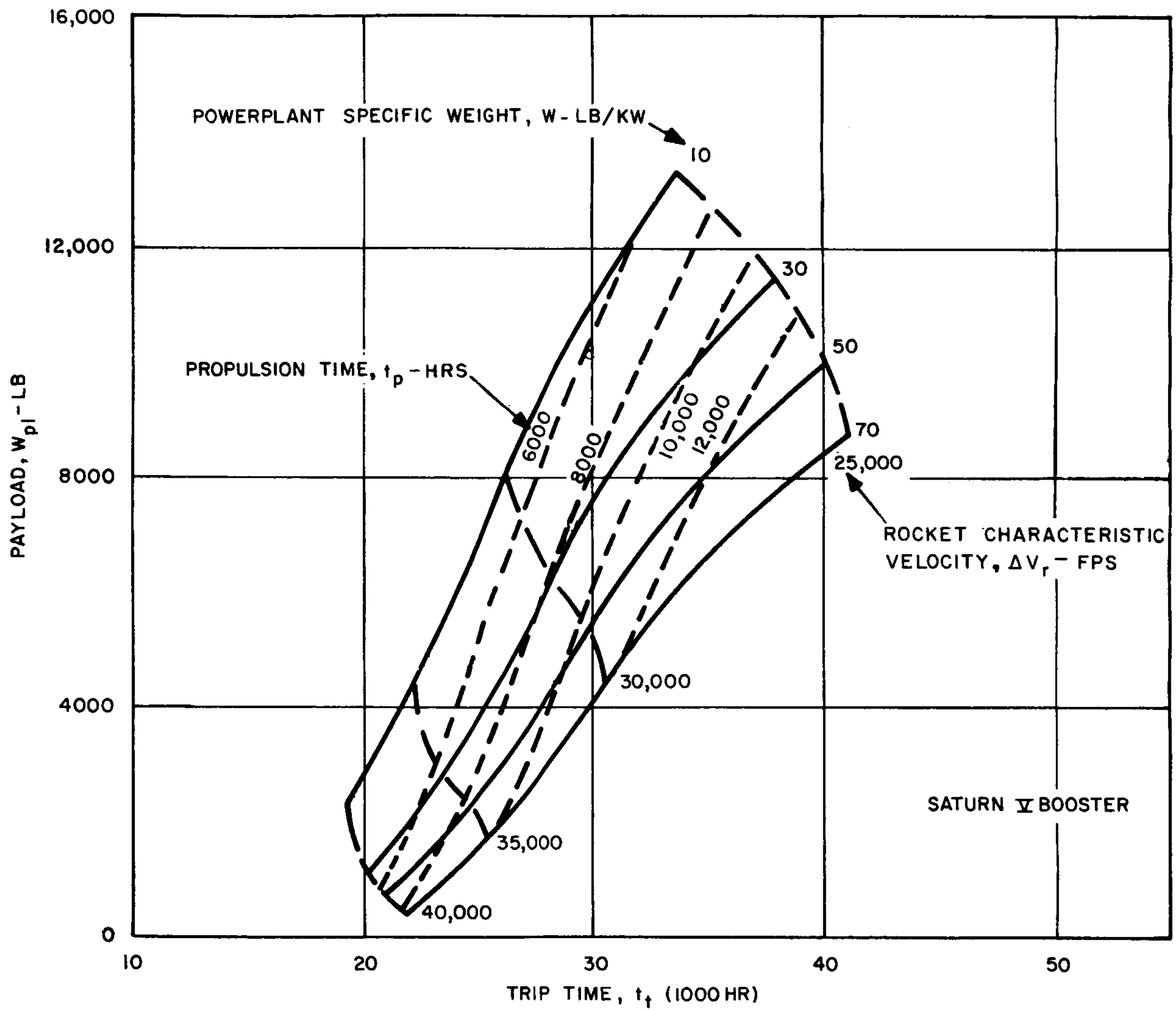


Figure 6.2-33. Saturn I Orbiter Performance

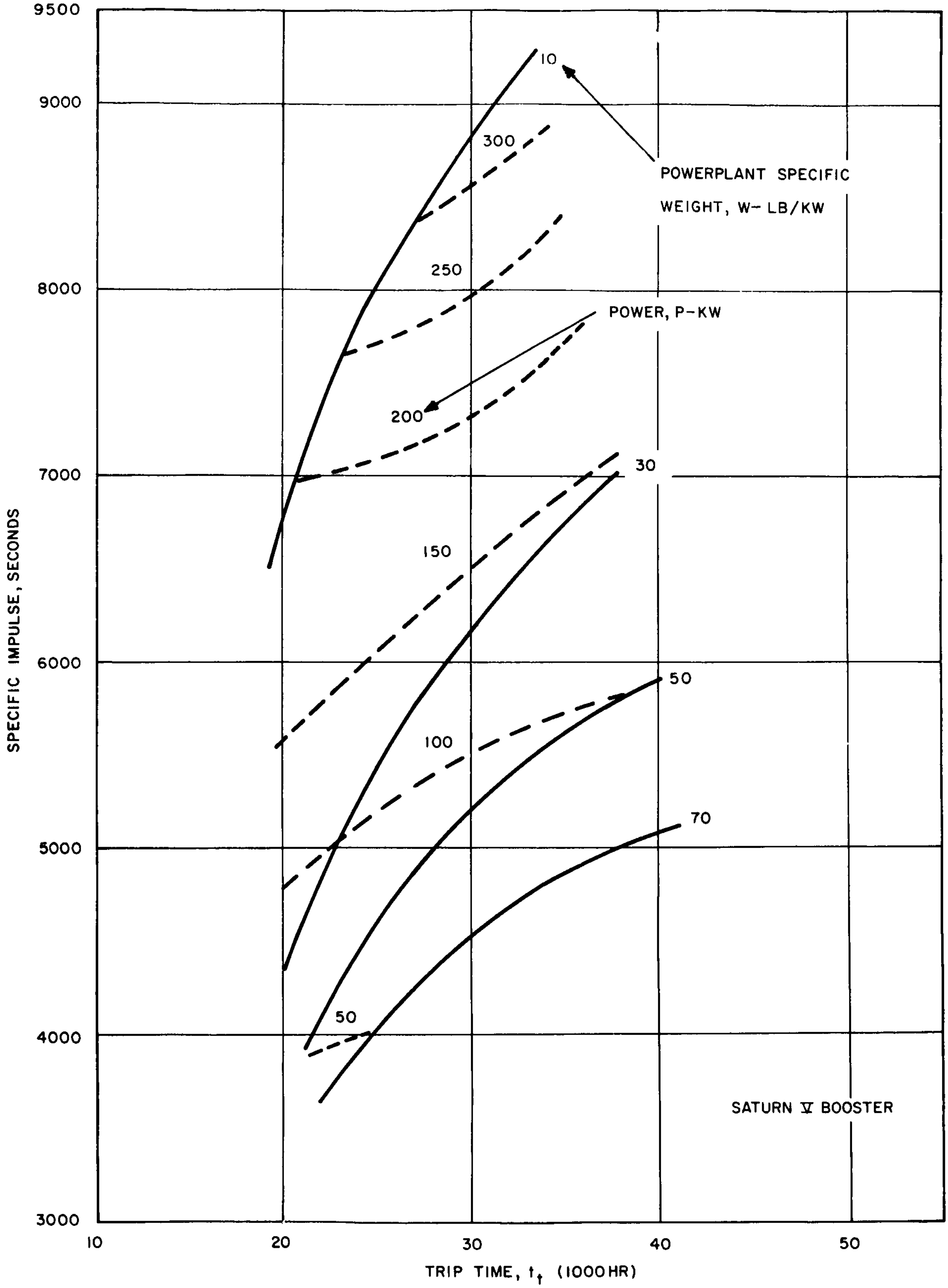


Figure 6.2-34. Saturn I Orbiter Requirements

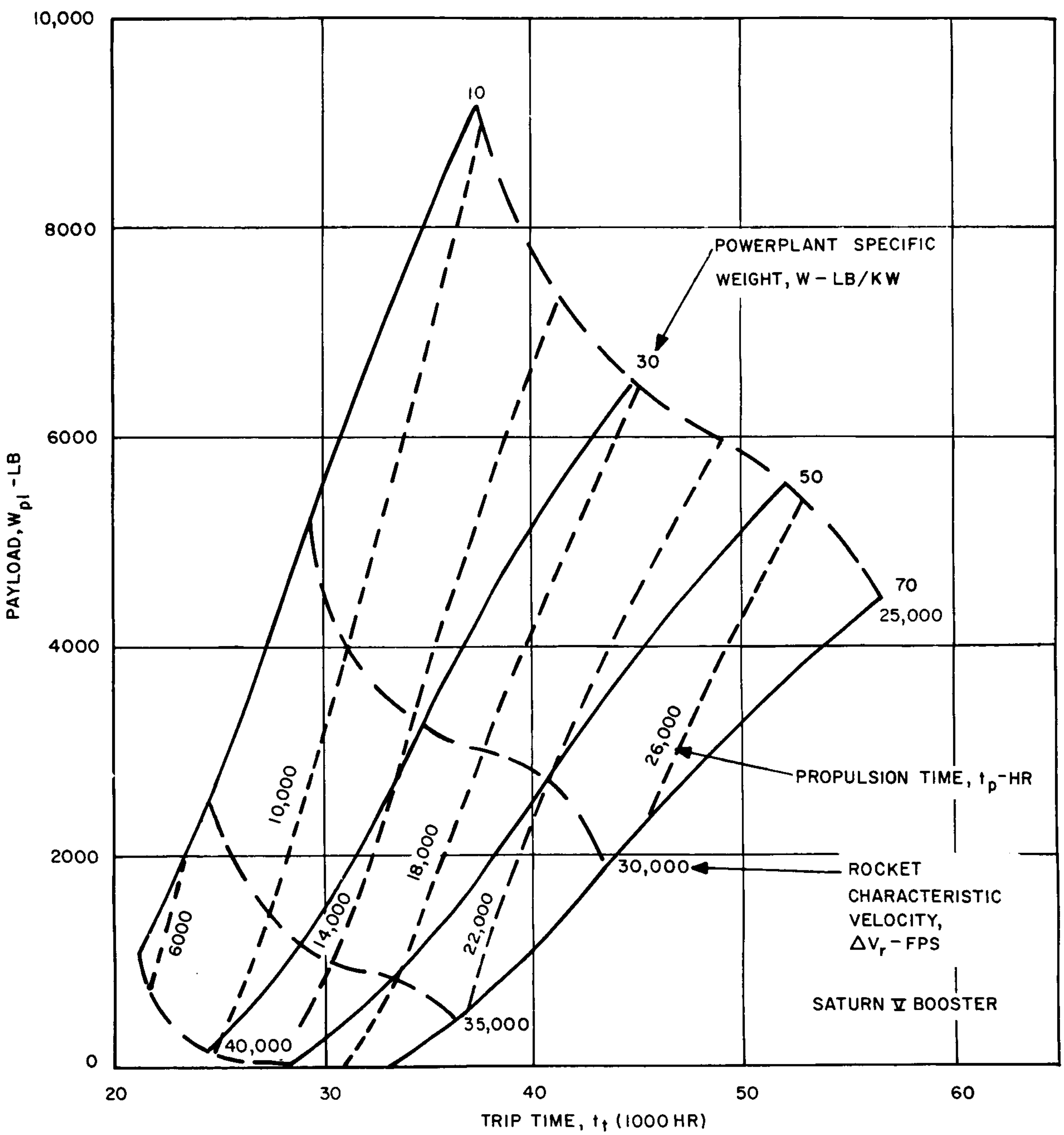


Figure 6.2-35. Saturn II Orbiter Performance

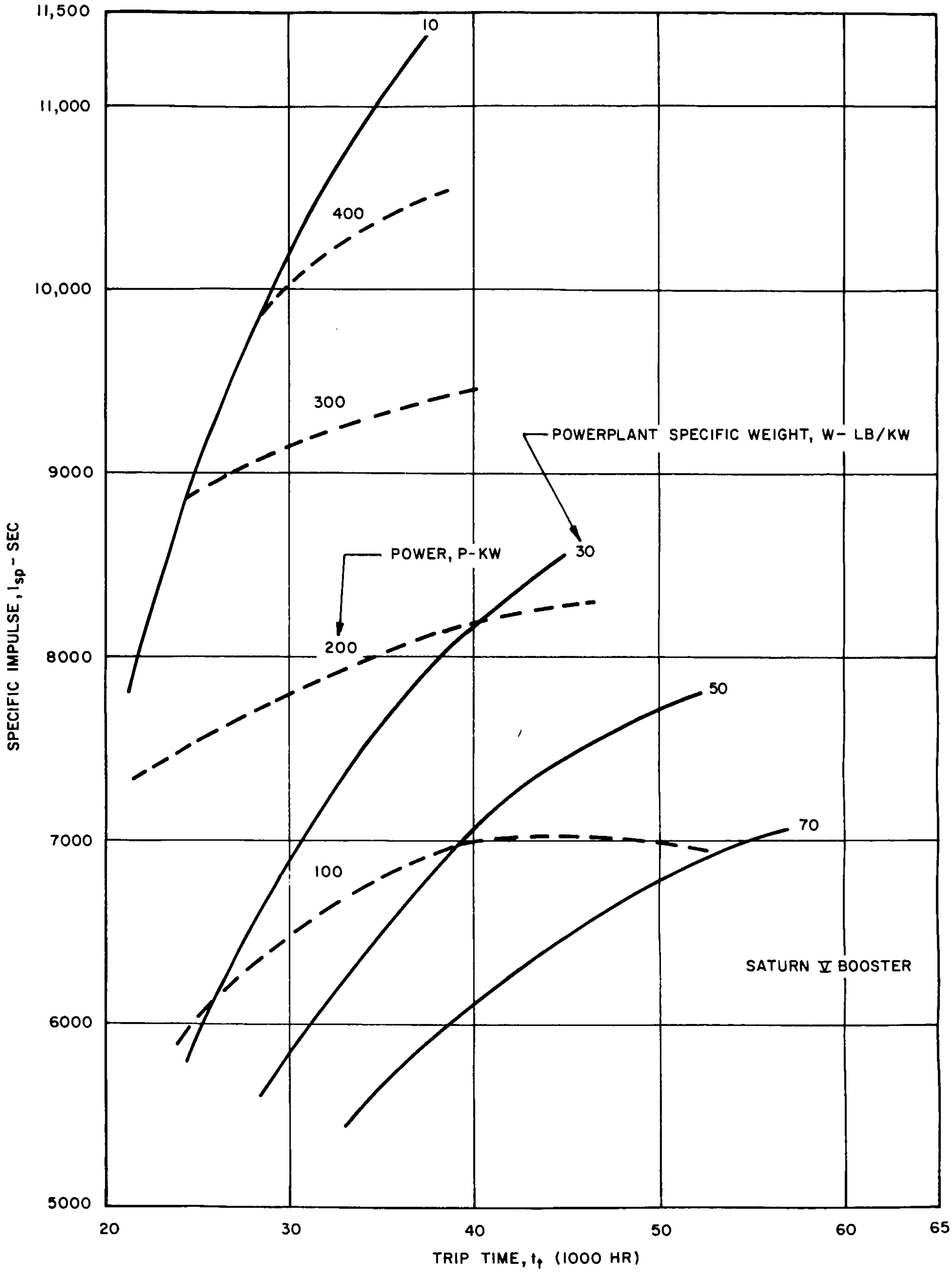


Figure 6.2-36. Saturn II Orbiter Requirements

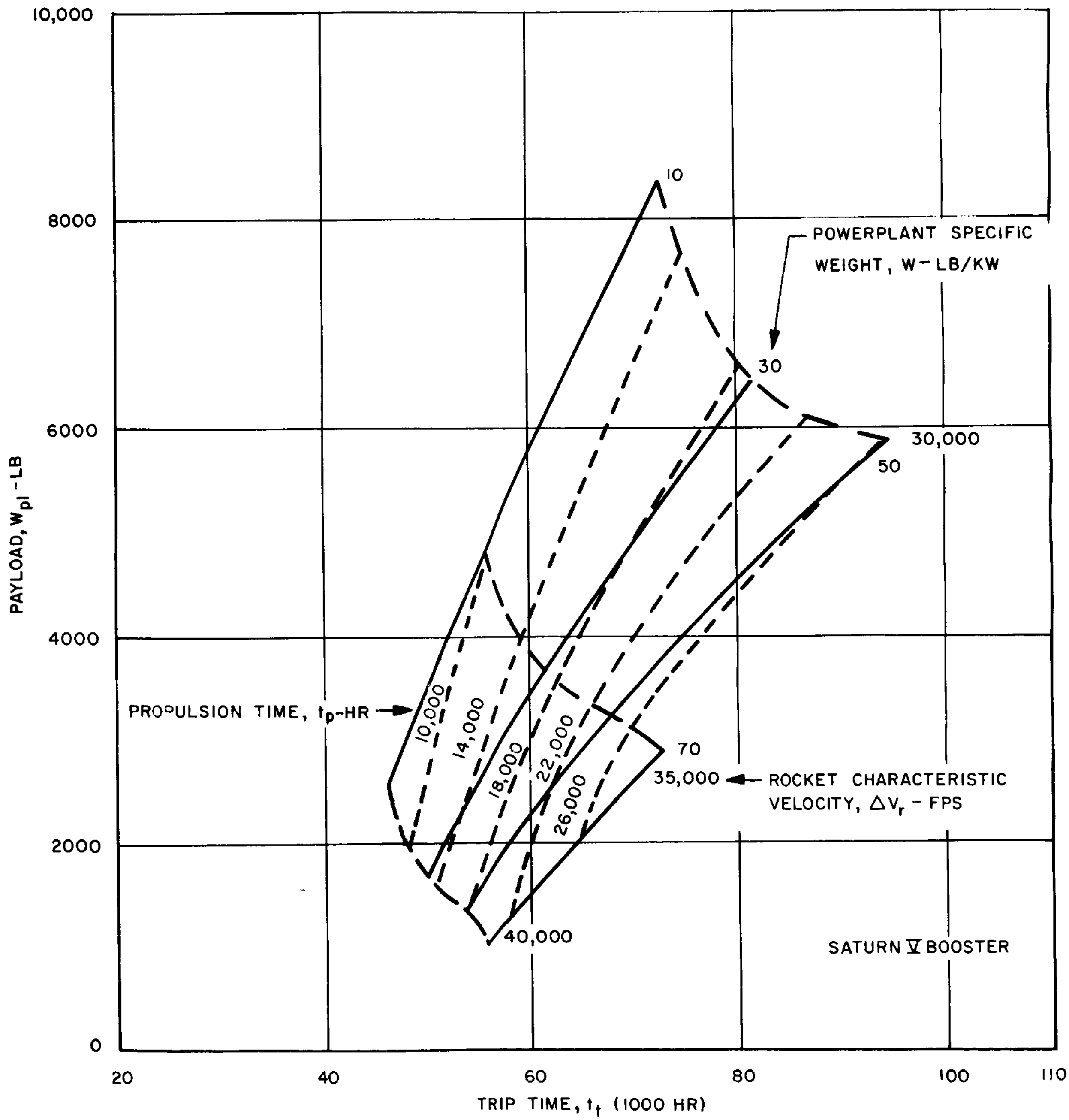


Figure 6.2-37. Uranus Orbiter Performance

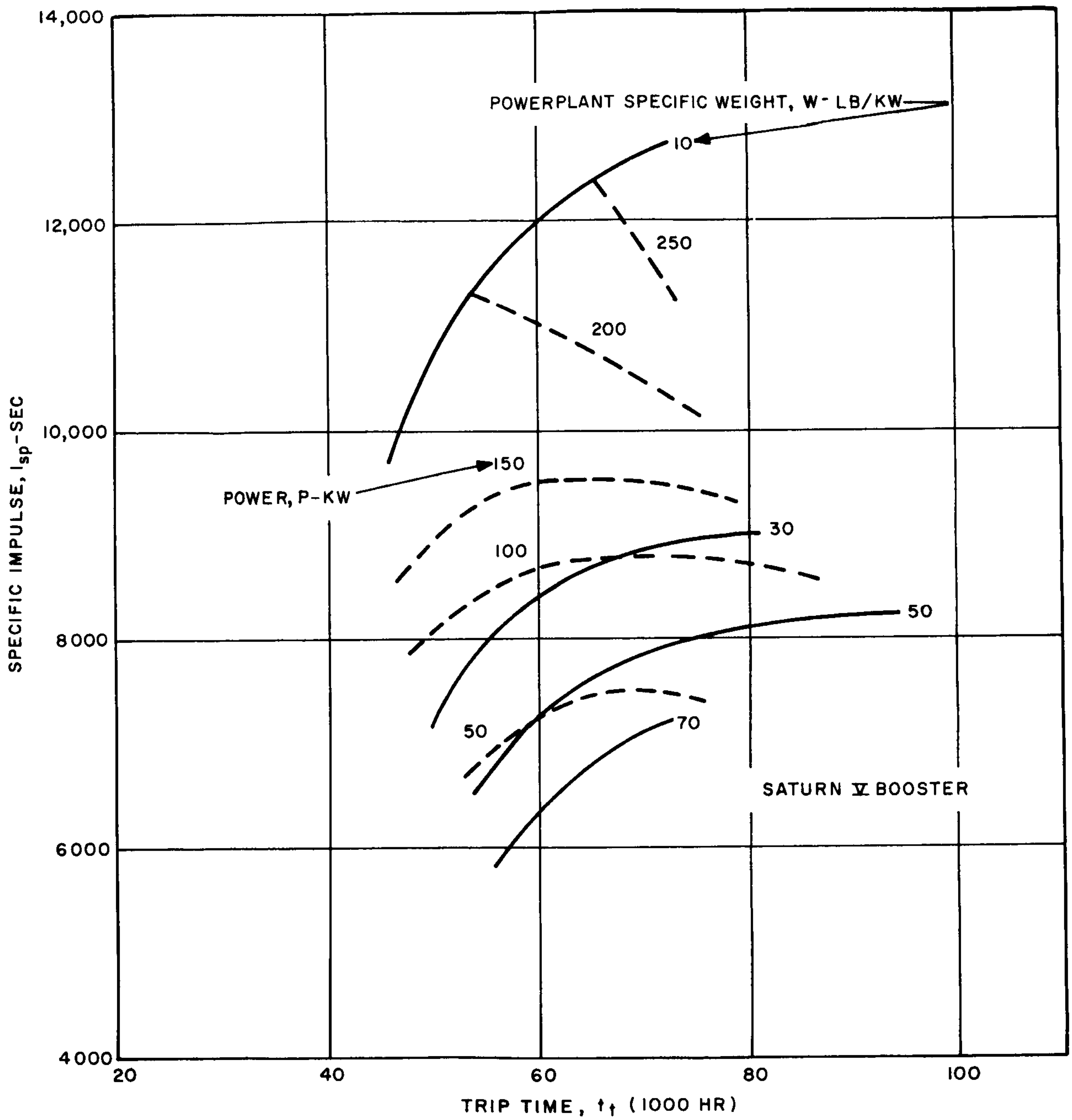


Figure 6.2-38. Uranus Orbiter Requirements

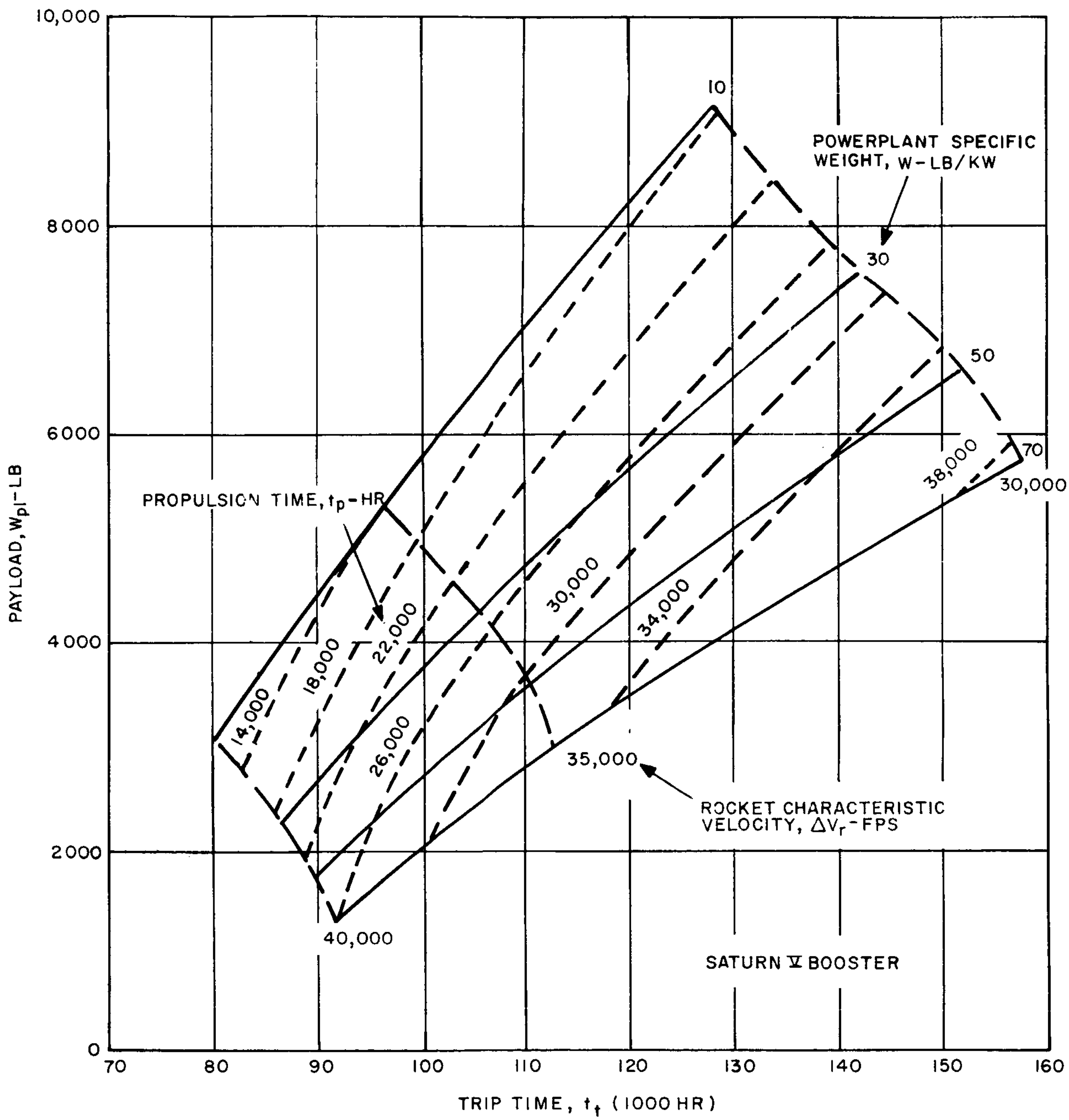


Figure 6.2-39. Neptune Orbiter Performance

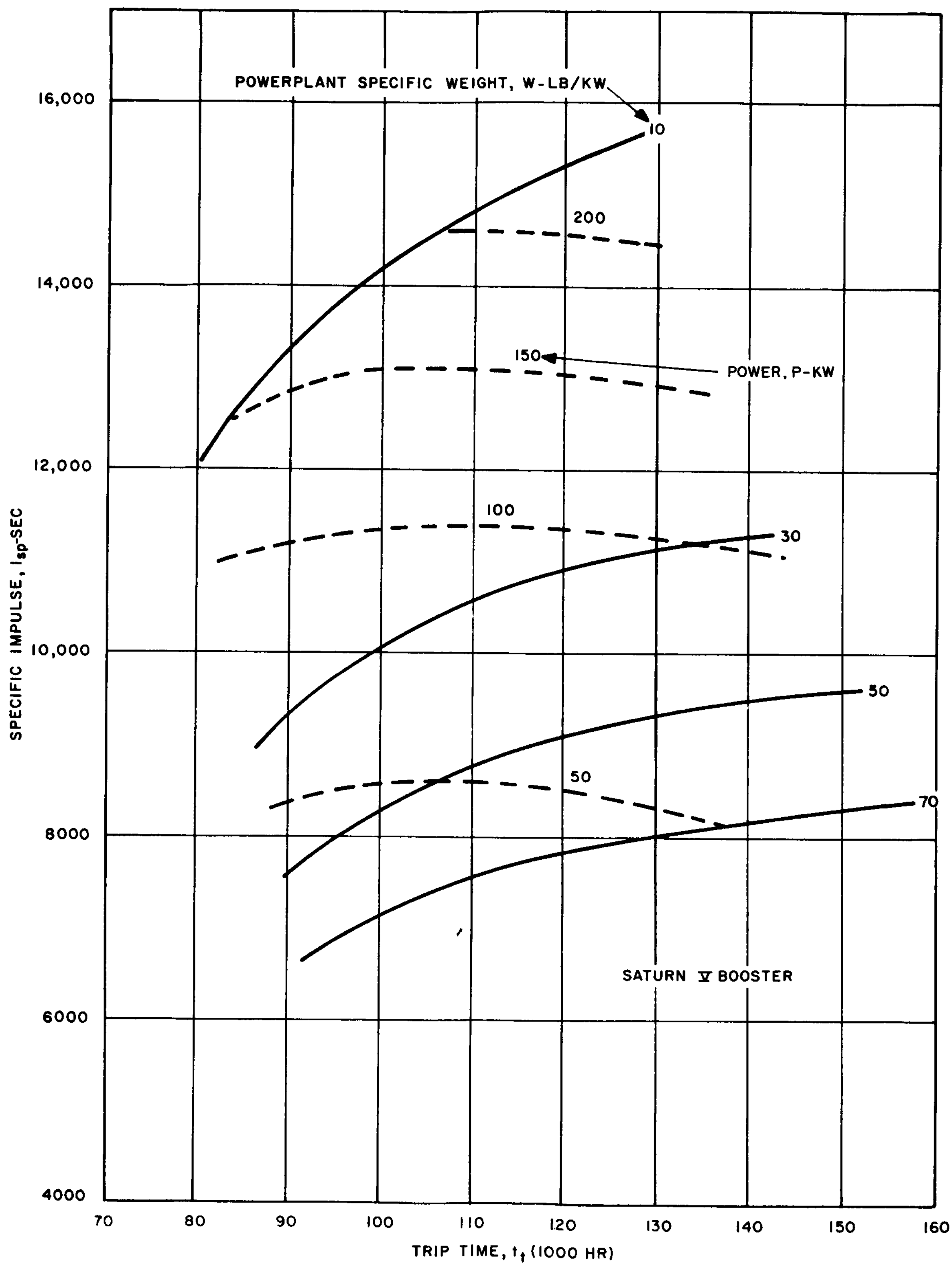


Figure 6.2-40. Neptune Orbiter Requirements

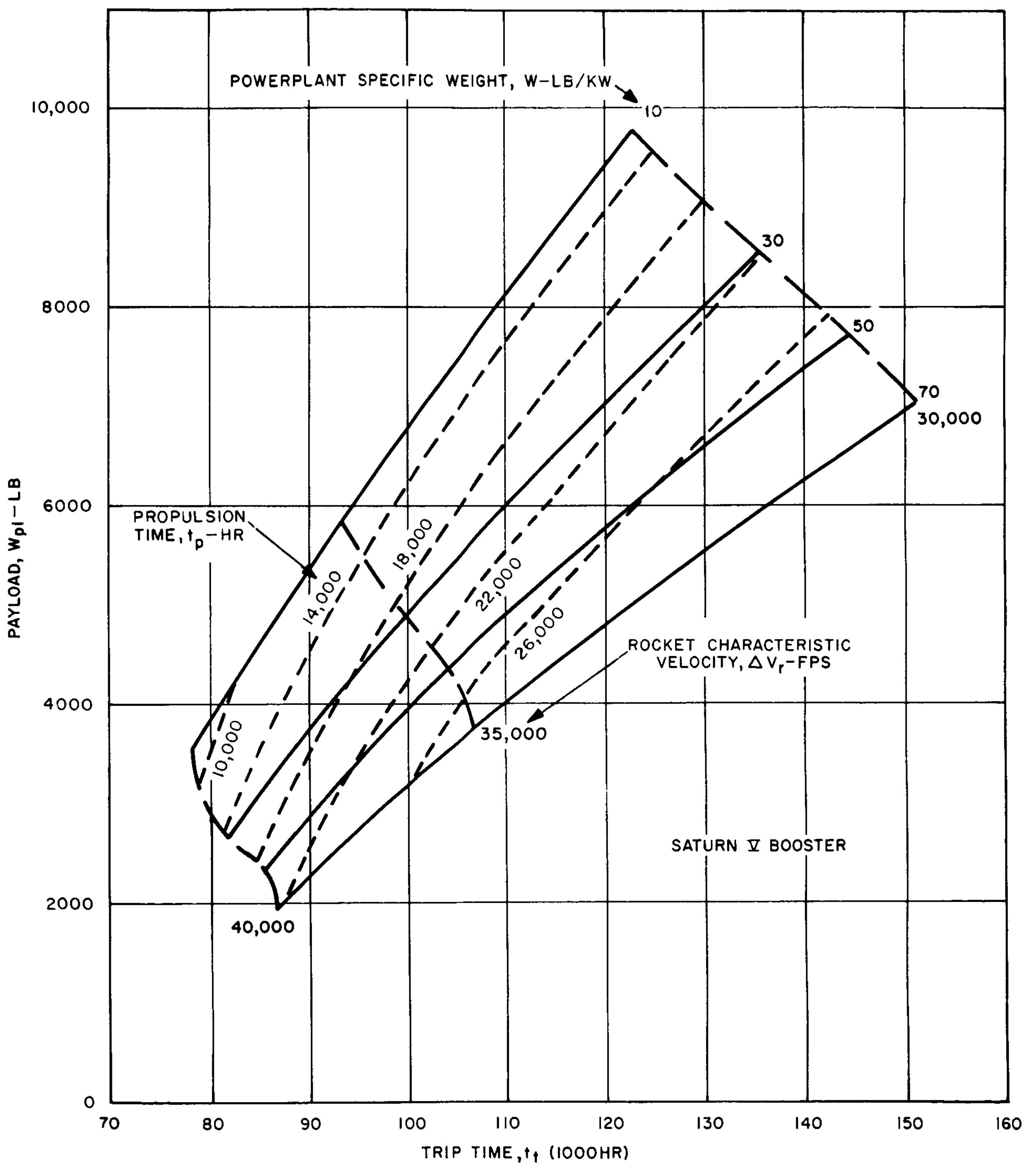


Figure 6.2-41. Pluto Orbiter Performance

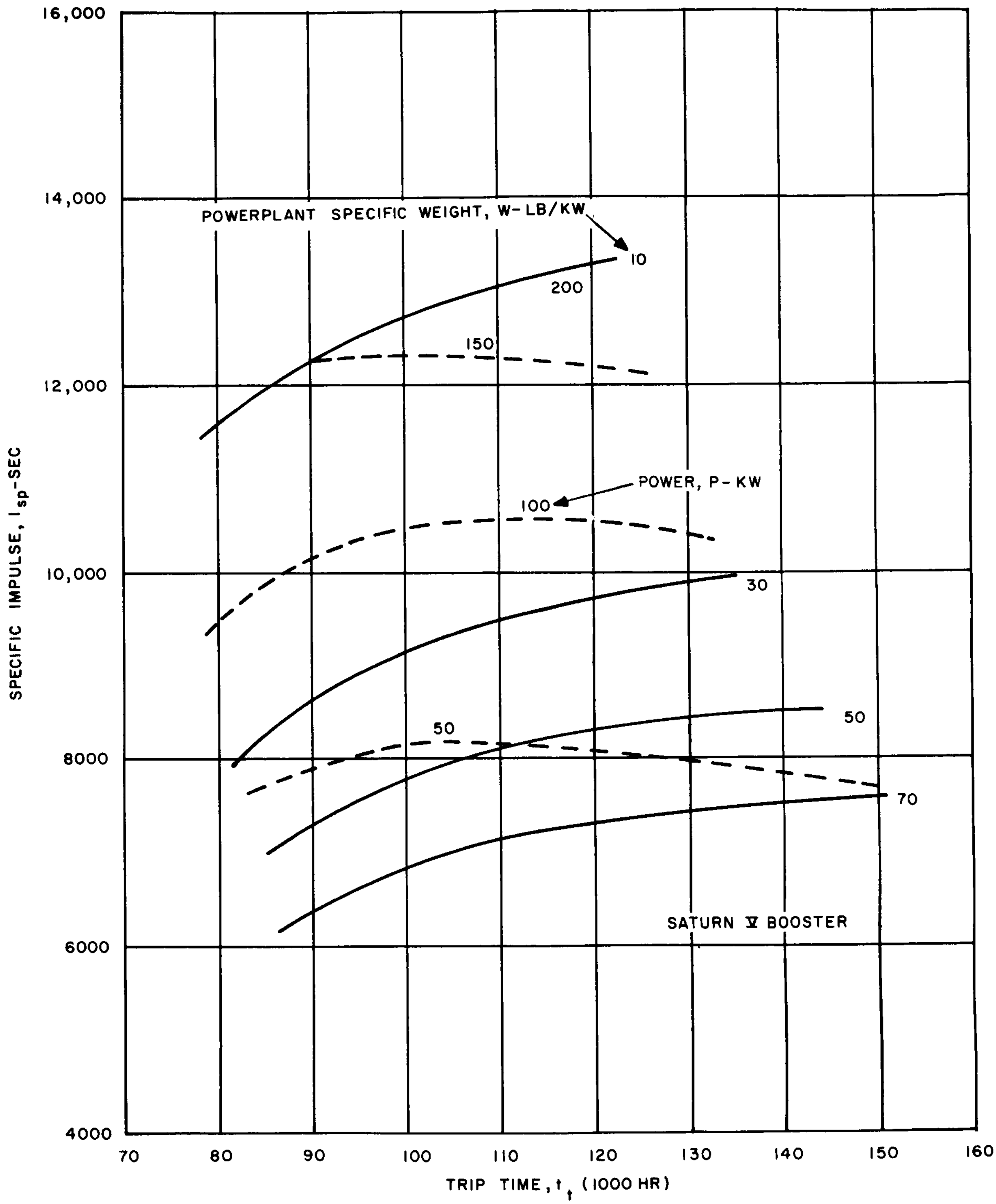


Figure 6.2-42. Pluto Orbiter Requirements

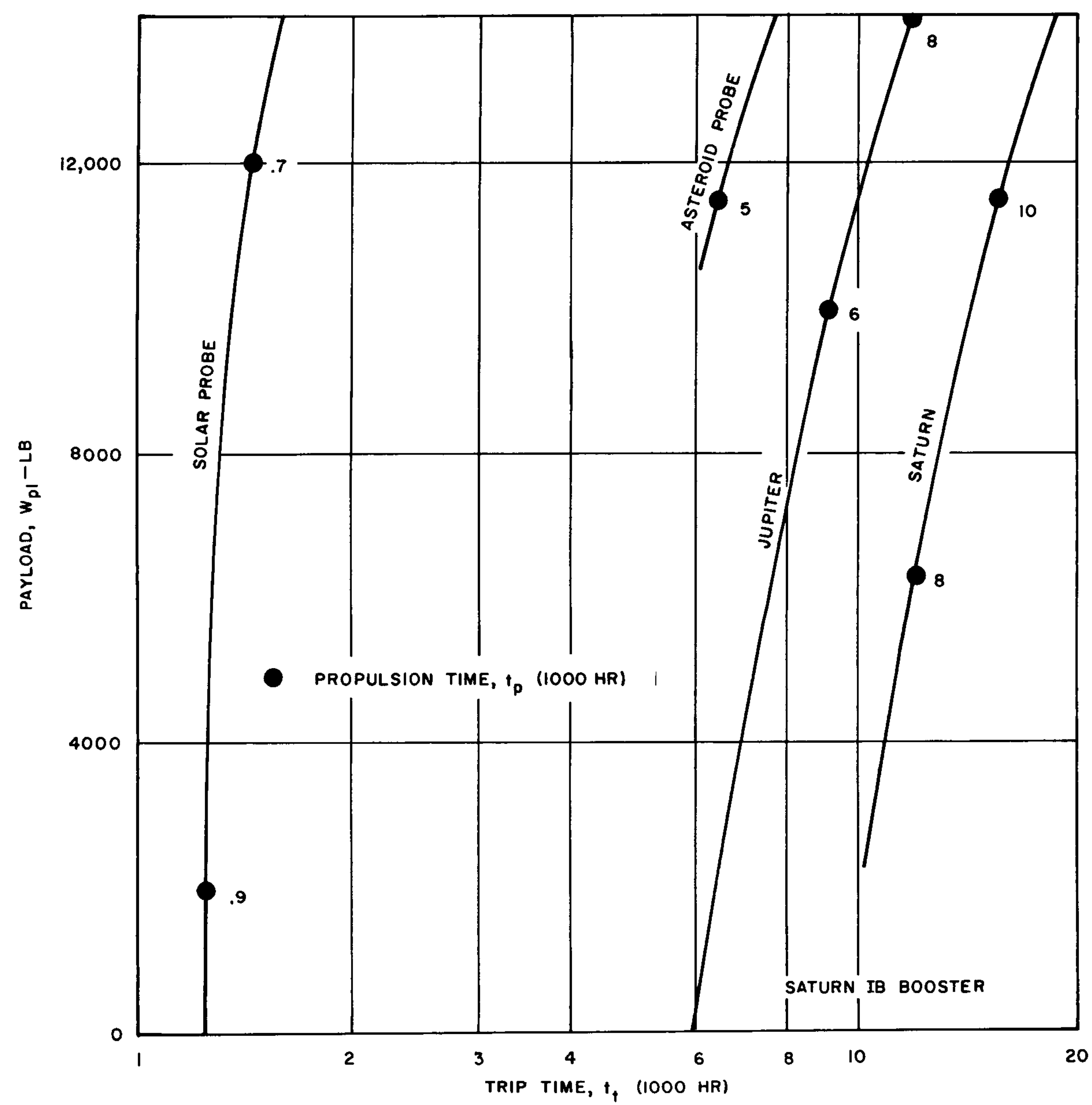


Figure 6.2-43. Fly-By Performance Summary - 10 Pounds per KW Powerplant - Saturn IB Booster

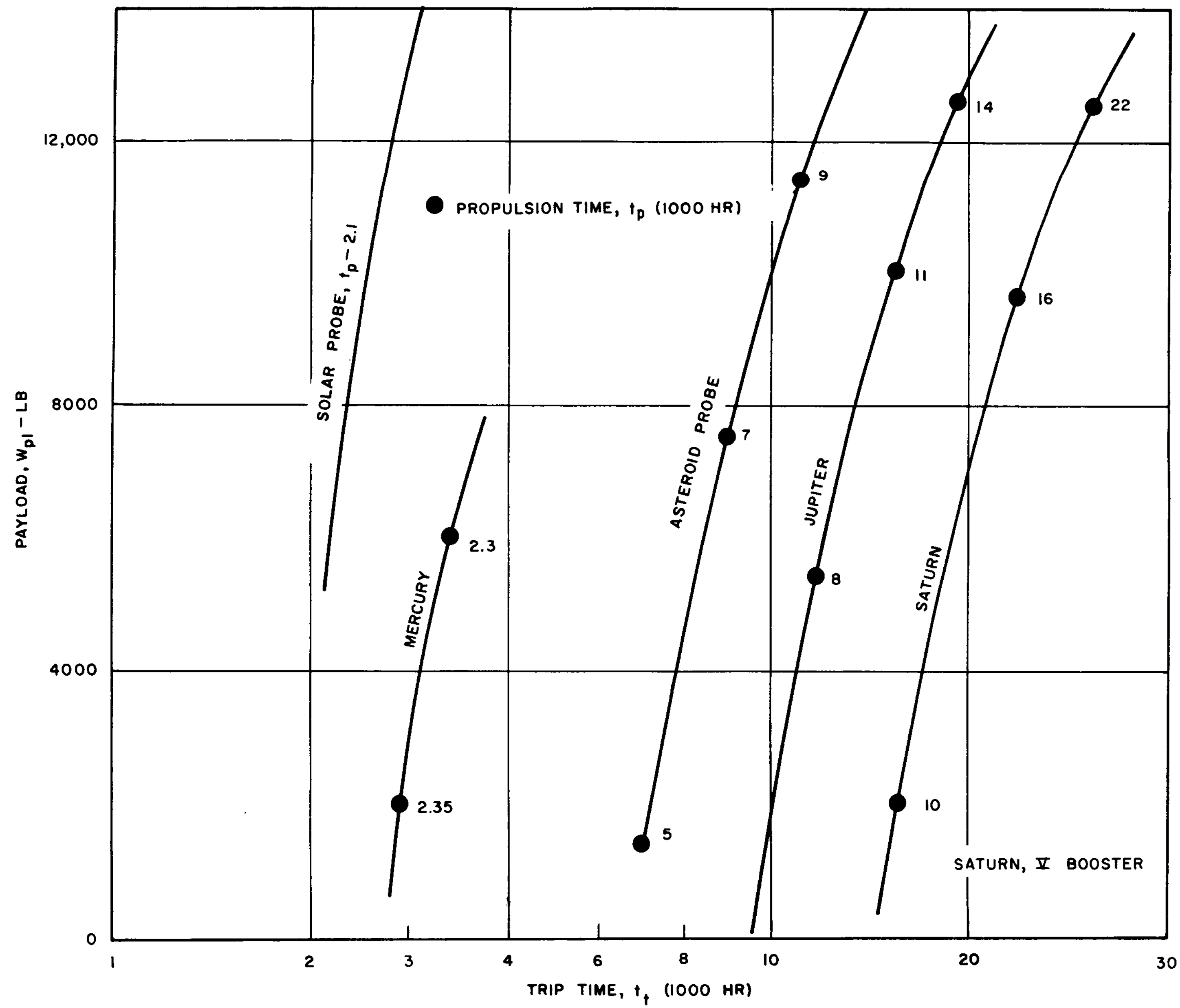


Figure 6.2-44. Fly-By Performance Summary - 30 Pounds per KW Powerplant - Saturn IB Booster

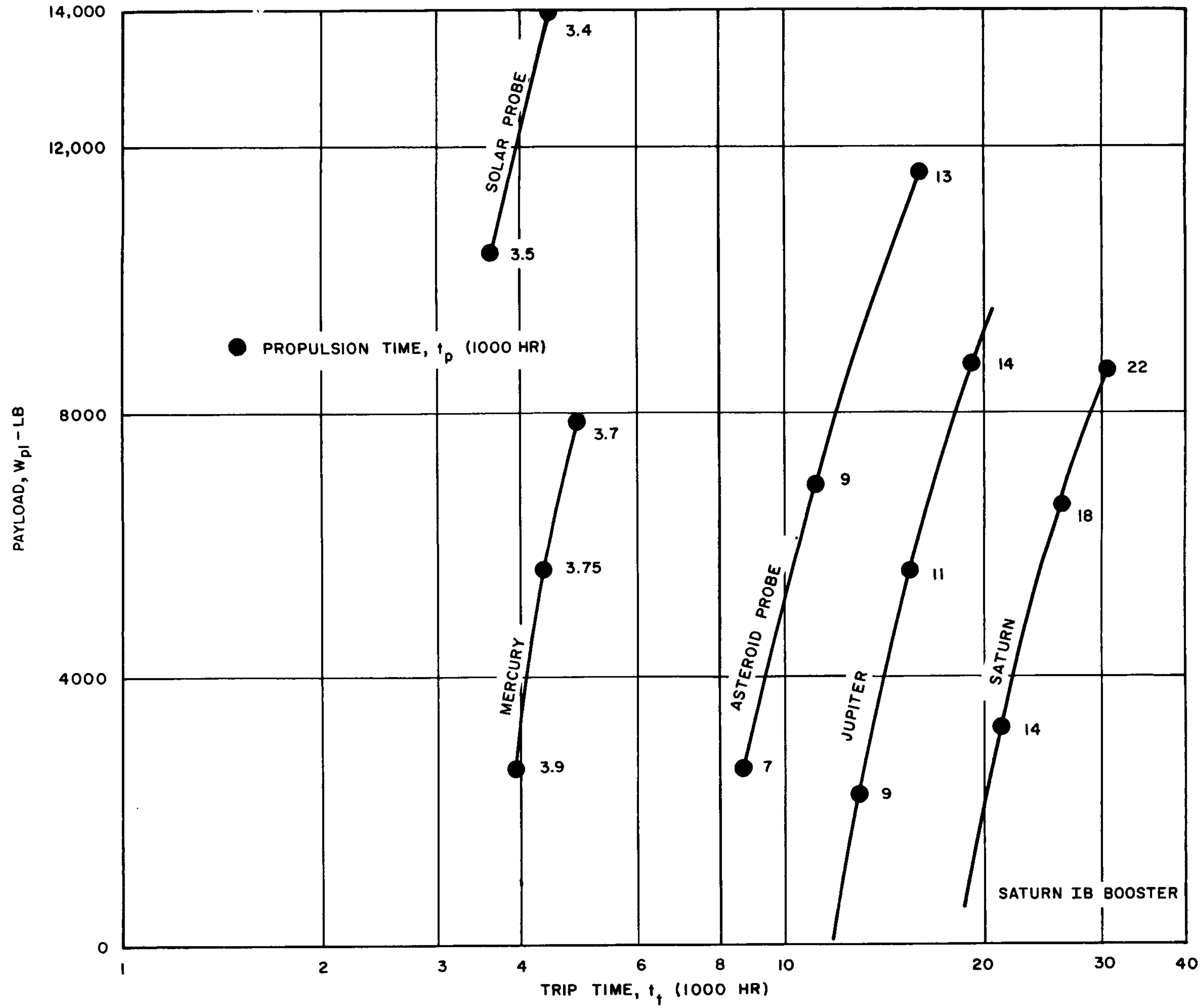


Figure 6.2-45. Fly-By Performance Summary - 50 Pounds per KW Powerplant - Saturn IB Booster

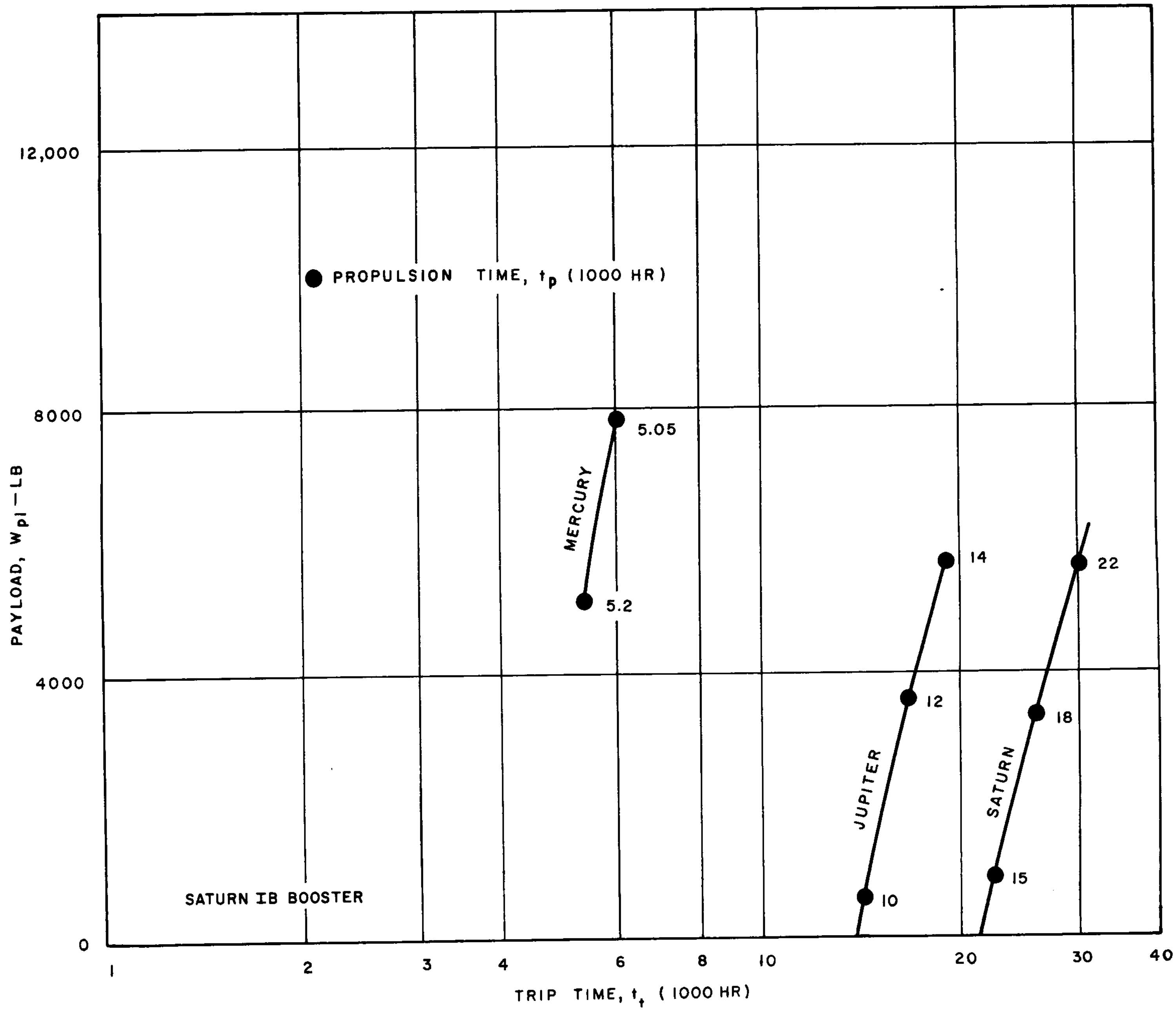


Figure 6.2-46. Fly-By Performance Summary - 70 Pounds per KW Powerplant - Saturn IB Booster

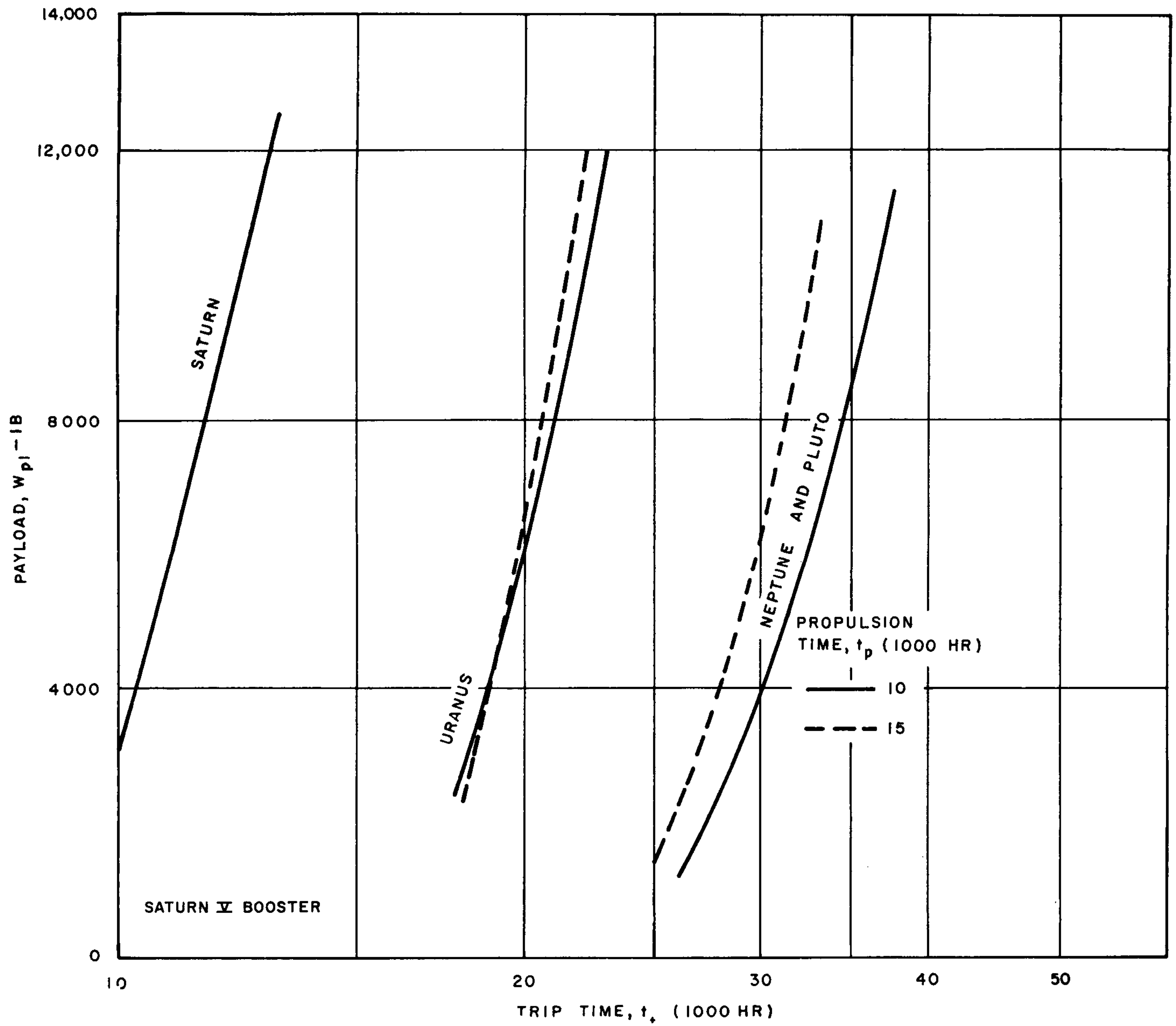


Figure 6.2-47. Fly-By Performance Summary - 30 Pounds per KW Powerplant - Saturn V Booster

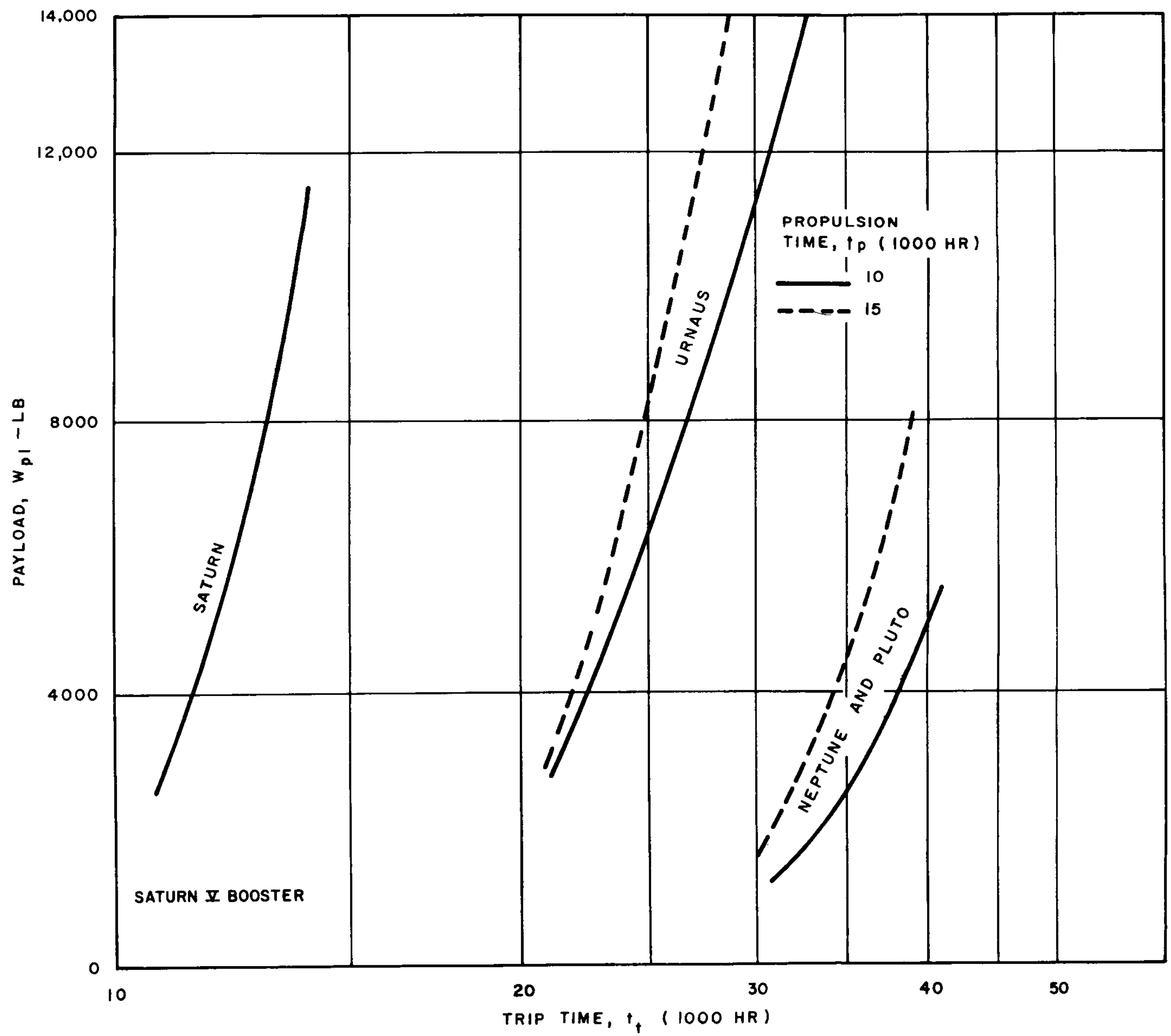


Figure 6.2-48. Fly-By Performance Summary - 50 Pounds per KW Powerplant - Saturn V Booster

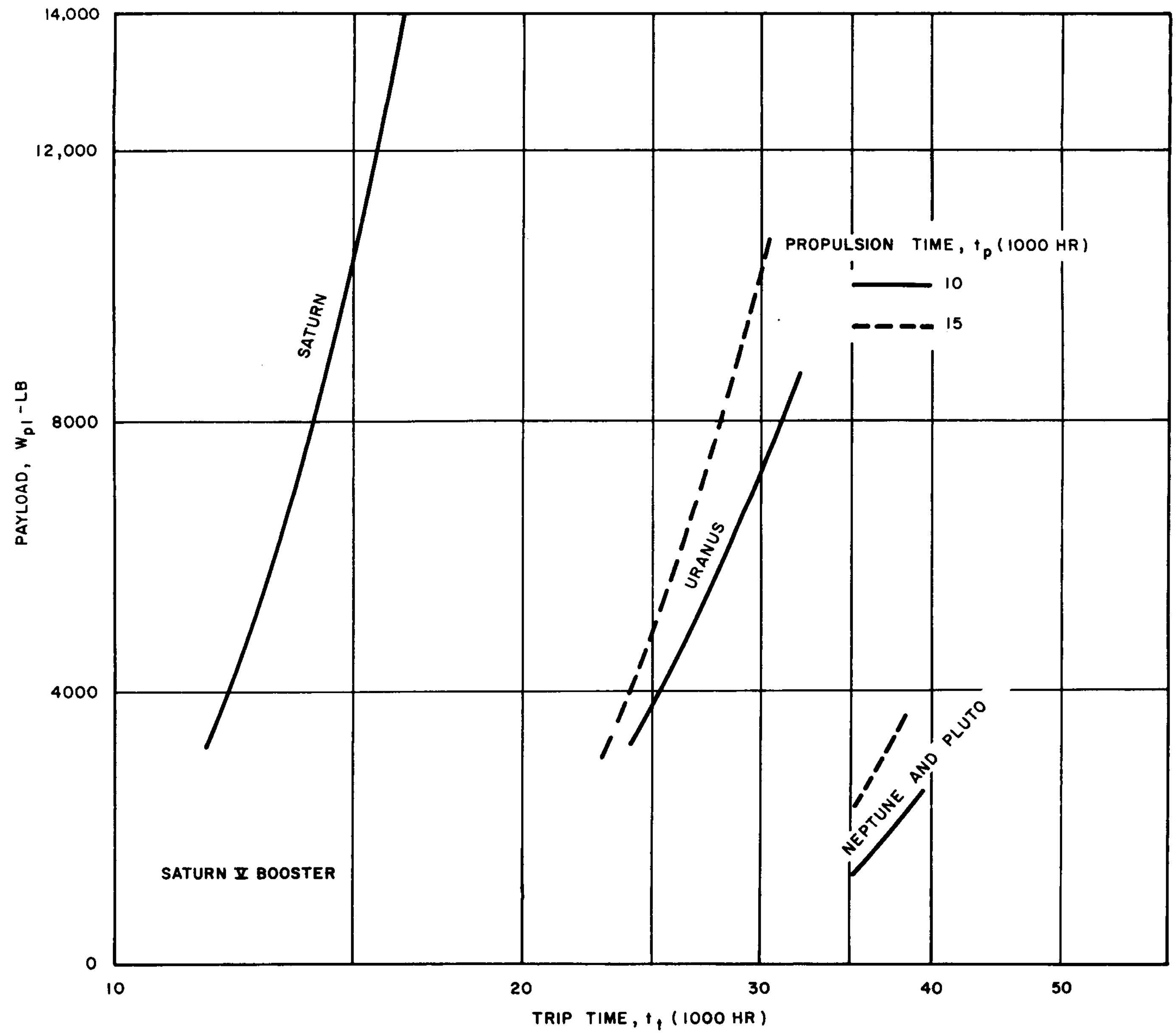


Figure 6.2-49. Fly-By Performance Summary - 70 Pounds per KW Powerplant - Saturn V Booster

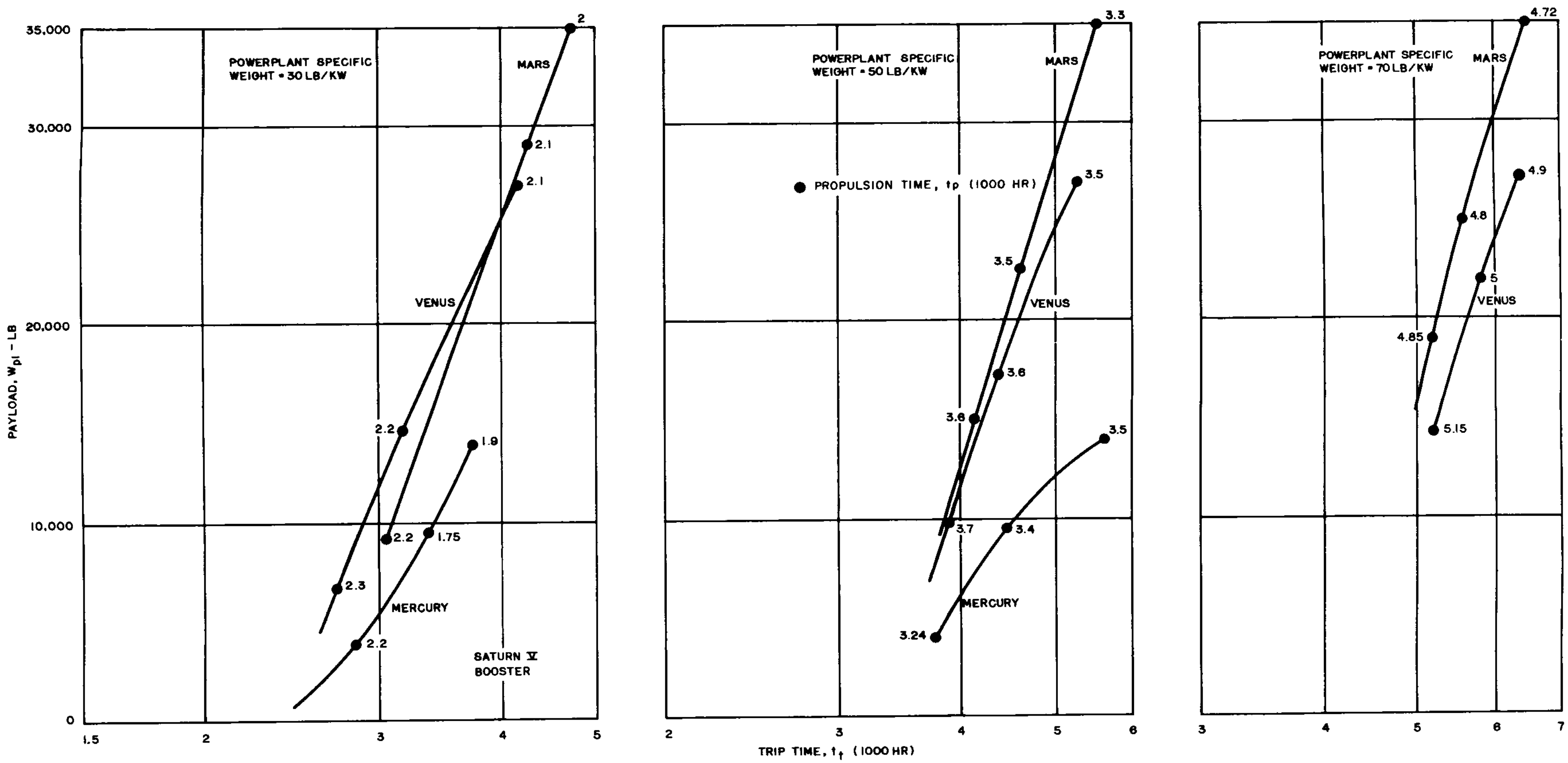


Figure 6.2-50. Minor Planet Orbiter Performance Summary

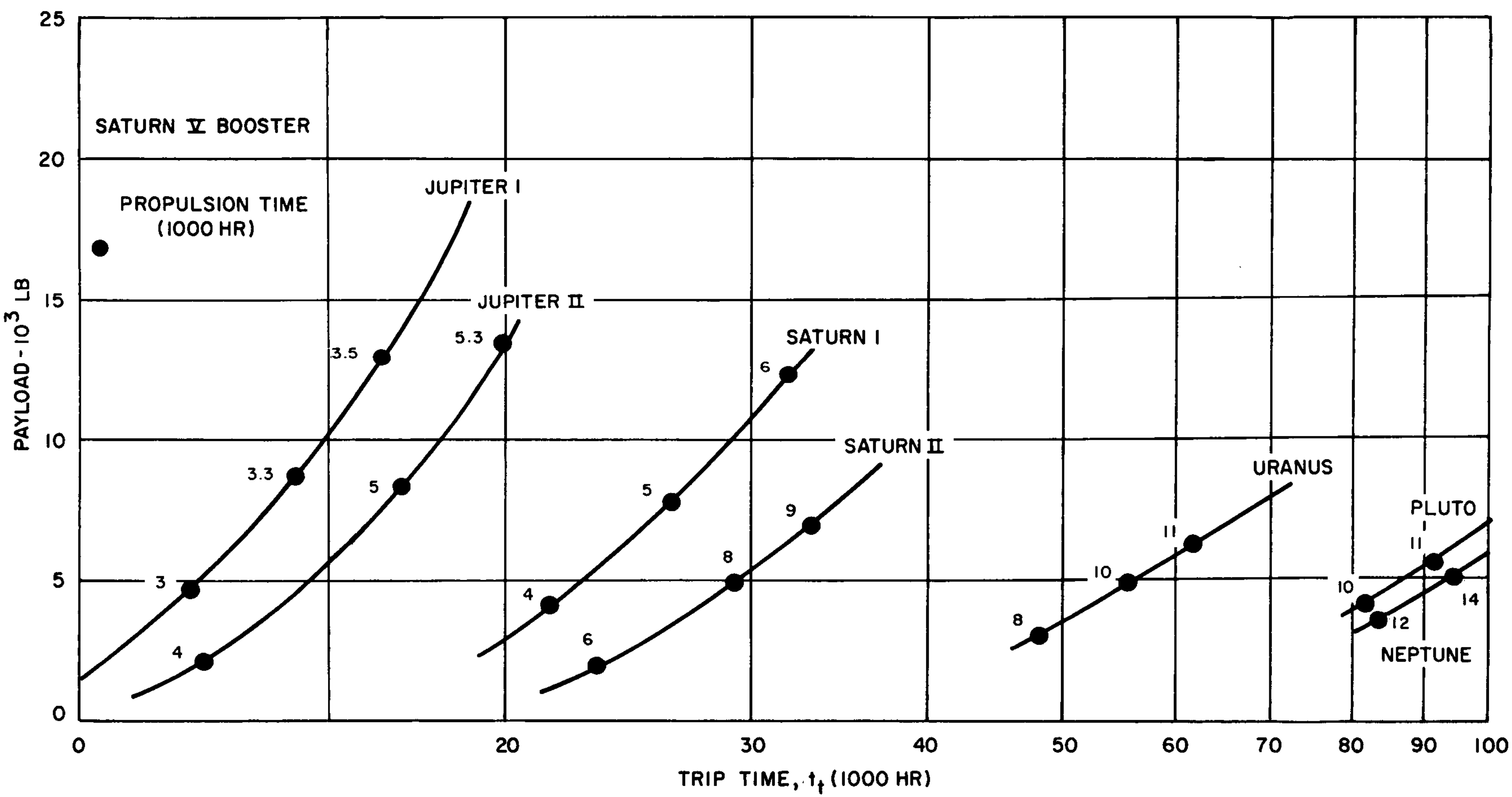


Figure 6.2-51. Major Planet Orbiter Performance Summary - 10 Pounds per KW Powerplant

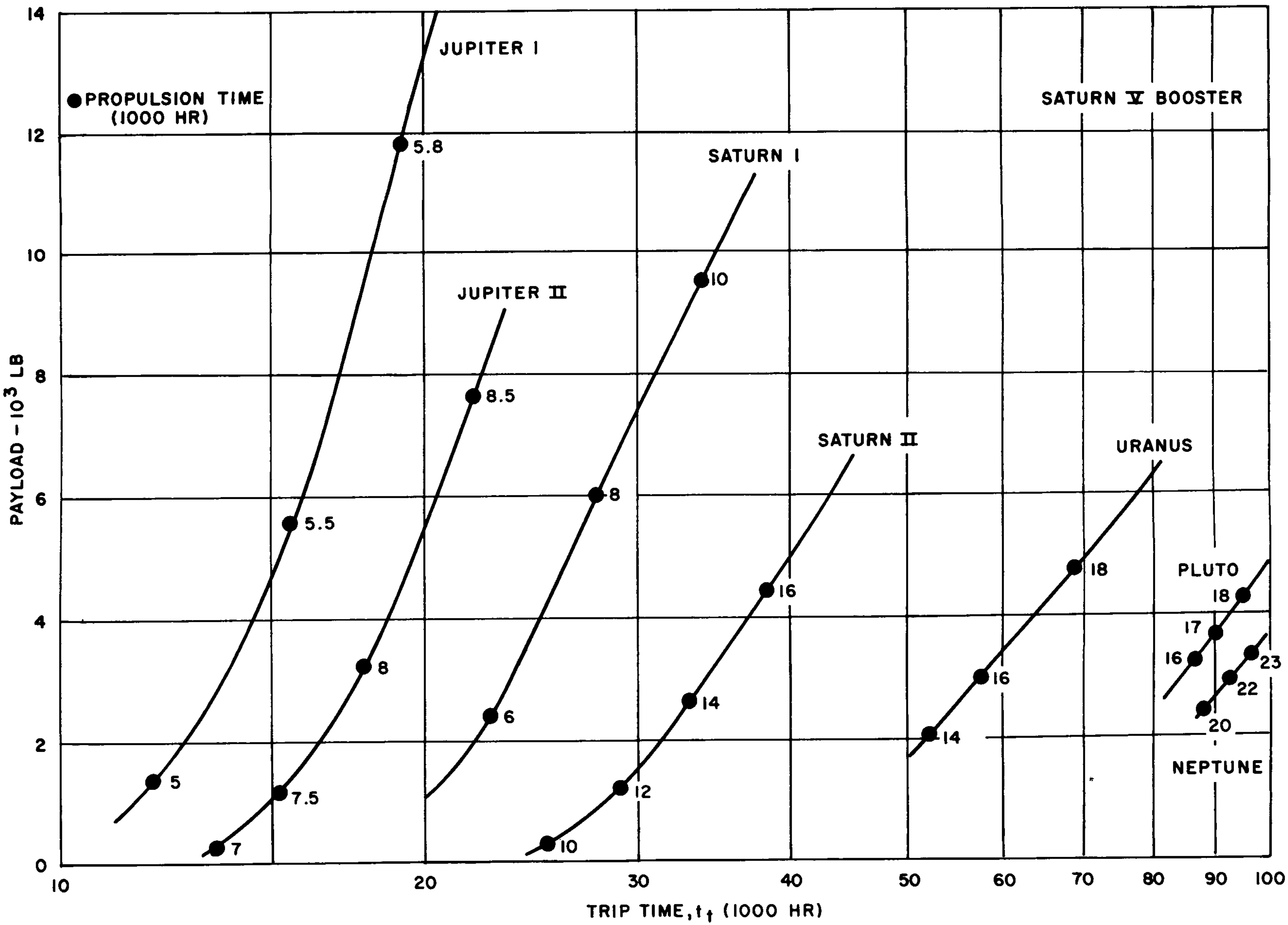


Figure 6.2-52. Major Planet Orbiter Performance Summary - 30 Pounds per KW Powerplant

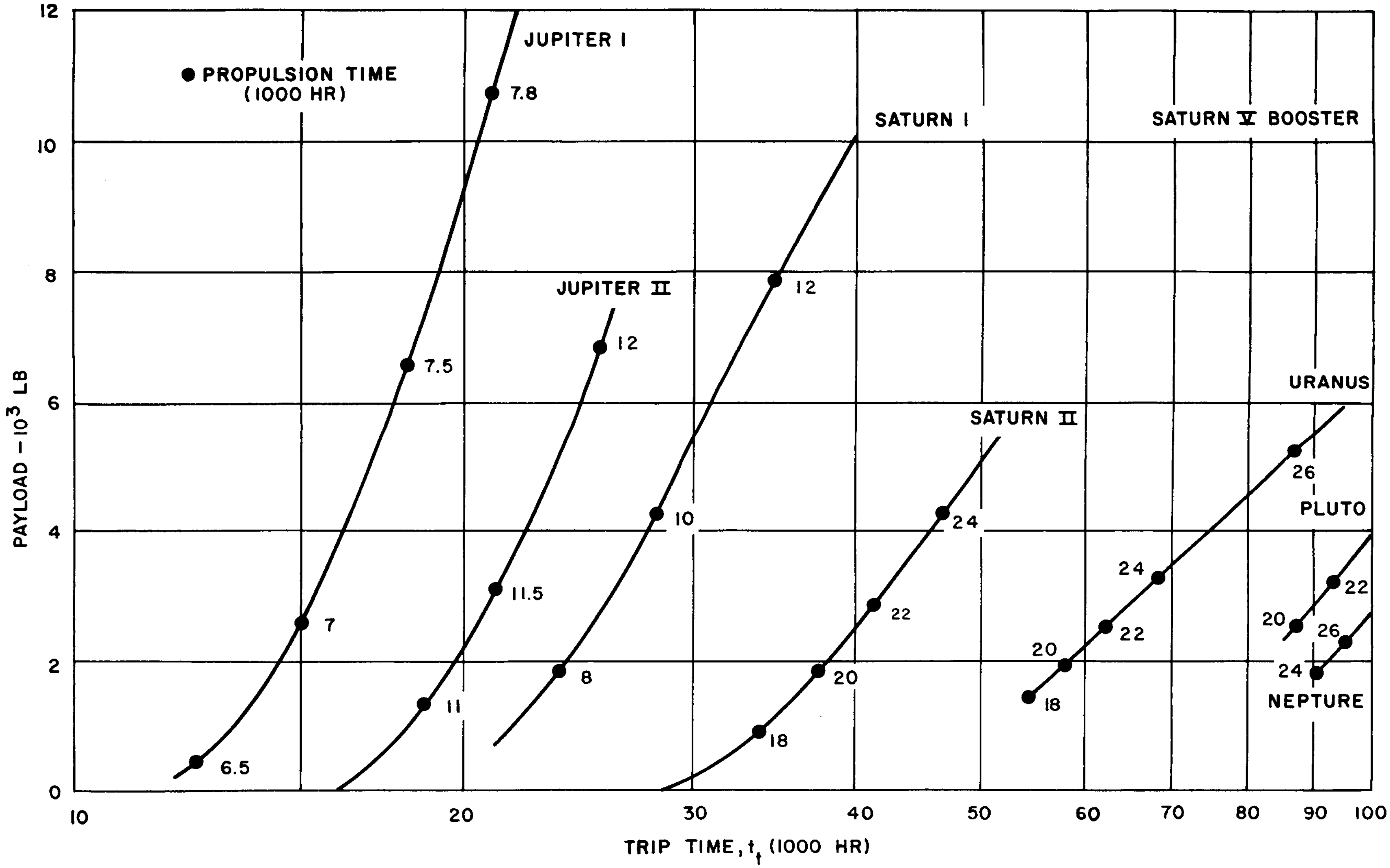


Figure 6.2-53. Major Planet Orbiter Performance Summary - 50 Pounds per KW Powerplant

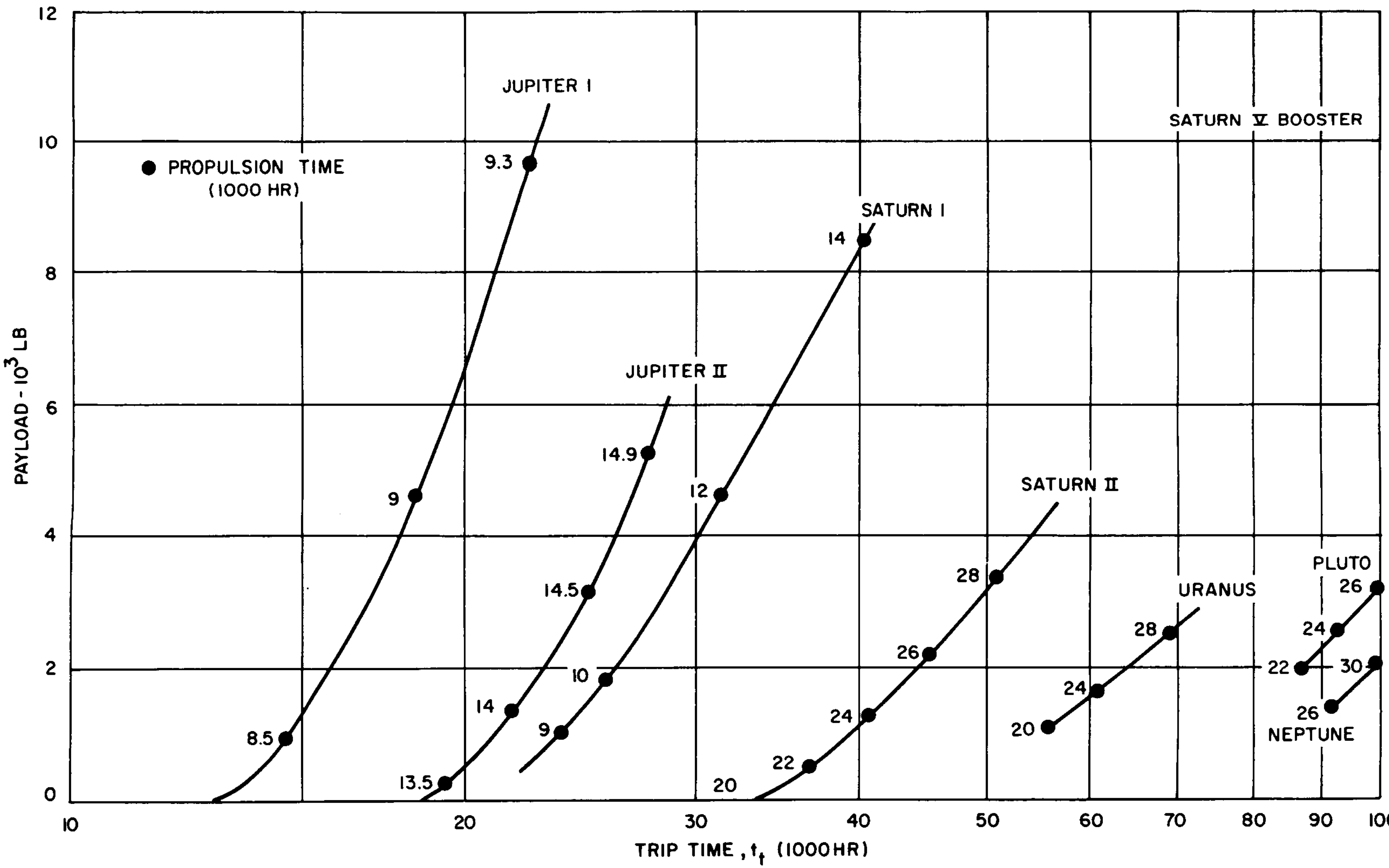


Figure 6.2-54. Major Planet Orbiter Performance Summary - 70 Pounds per KW Powerplant

7. VARIABLE SPECIFIC IMPULSE

The variable thrust (variable specific impulse) propulsion requirements illustrated in Figure 4.2-1 were obtained from a series of optimum power-limited trajectories calculated by the variational calculus. These trajectories were constrained to operate at constant power throughout the trajectory but were completely unconstrained with respect to the thrust and specific impulse levels. Comparisons of the variable specific impulse requirements with similar trajectory results constrained to operate at constant or zero thrust indicate a reduction in the propulsion parameter J on the order of 15 percent for the variable specific impulse case at constant trip time. The specific impulse variation required to achieve this reduction in J is, however, of the order of 30:1 to 40:1.

The objective of this phase of the investigation was, therefore, to determine if a substantial fraction of the theoretical performance improvement can be obtained with a moderate degree of specific impulse variation that can possibly be designed into a first or second generation ion propulsion system. The optimum variable specific impulse schedule can be approximated by assuming a linear variation in acceleration with time.

$$a = a_o \left[1 - b\,t \right] \tag{7.1}$$

Equation 7.1 can then be used to modify the conventional relationships for characteristic length (Table 4.5-1), optimum initial acceleration (Equation 6.5), and mass ratio (Equation 6.6).

For the orbiters, the one dimensional velocity profile is like that shown in Figure 3-1 where during the low thrust phase the profile is a quadratic given by the integral of Equation 7.1.

$$V = V_o - a_o t\,(1 - bt/2) \tag{7.2}$$

Since the terminal velocity is zero,

$$V_o = a_o t_{ph} (1 - bt_{ph}/2) \tag{7.3}$$

The characteristic length is defined by the integral of Equation 6.2.

$$L = V_o t_h - a_o t_{ph}^2/2 + ba_o t_{ph}^3/6 \tag{7.4}$$

Substituting Equation 7.3 and solving for the initial acceleration yields

$$a_o = \frac{L}{t_{ph} \left[t_h - t_{ph}/2 - bt_{ph} (3t_h - t_{ph})/6 \right]} \tag{7.5}$$

where the characteristic length is determined from the quadratic relationship of Figure 4.3-1.

In order to obtain an expression for mass ratio, the equation defining specific impulse is integrated.

$$I_{sp} = - \frac{T}{dW/dt} = - \frac{aW/g}{dW/dt} \tag{7.6}$$

The variation of power to thrust ratio with initial specific impulse is the same as that of the constant thrust engine, although for the integration, the following approximation is made:

$$P/T = A_o + A_1 I_{spo} \approx A_1 I_{spo} \tag{7.7}$$

Substituting Equations 7.1 and 7.7 into 7.6 yields

$$dW/W^2 = - (A_1 a_o^2/g^2 P) (1 - bt)^2 dt \tag{7.8}$$

Integration of Equation 7.8 at constant power results in

$$1/W_o - 1/W = -(A_1 a_o^2/g^2 P)\left[1 - bt_{ph} + (bt_{ph})^2/3\right] t_{ph} \tag{7.9}$$

Substituting for power from Equation 7.7 and rearranging, yields the following expression for mass ratio:

$$\mu = \frac{W}{W_o} = \frac{g\,I_{spo}}{g\,I_{spo} + a_o t_{ph}\left[1 - bt_{ph} + (bt_{ph})^2/3\right]} \tag{7.10}$$

The optimum initial specific impulse is determined by maximizing the general expression for payload given by Equation 6.1. Substituting Equations 6.3, 6.7, and 7.10 into 6.1 results in

$$\frac{W_{pl}}{W_o} = \left\{\frac{g\,I_{spo}\,(1 + w_t)}{g\,I_{spo} + a_o t_{ph}\left[1 - bt_{ph} + (bt_{ph})^2/3\right]} - w_t - \frac{wa_o\,(A_o + A_1\,I_{spo})}{g}\right\} \tag{7.11}$$

Differentiating with respect to initial specific impulse at constant propulsion time and initial acceleration,

$$\frac{1}{W_o}\left.\frac{\partial W_{pl}}{\partial I_{spo}}\right|_{t_{ph},\,a_o} = \frac{g\,t_{ph}\,(1 + w_t)\left[1 - bt_{ph} + (bt_{ph})^2/3\right]}{\left\{g\,I_{spo} + a_o t_{ph}\left[1 - bt_{ph} + (bt_{ph})^2/3\right]\right\}^2} - \frac{W\,A_1}{g} \tag{7.12}$$

Equating 7.12 to zero and solving for the initial specific impulse results in the following:

$$I_{spo} = \frac{a_o t_{ph}}{g}\left[1 - bt_{ph} + (bt_{ph})^2/3\right] + \left\{\frac{t_{ph}\,(1 + w_t)}{A_1\,w}\left[1 - bt_{ph} + (bt_{ph})^2/3\right]\right\}^{0.5} \tag{7.13}$$

Equations 7.10 and 7.13 are valid for the fly-by case, but a slightly different equation than 7.5 must be used for the initial acceleration. The fly-bys allow an additional degree of freedom over the orbiters since the planetary propulsion constraint is relaxed. Hence, the hyperbolic excess velocity cannot be calculated as in Equation 7.3 but must be arbitrarily specified. Integrating the acceleration equation twice and solving for the initial acceleration yields the following:

$$a_o = \frac{L - V_o t_h}{t_{ph}^{2}/2\,(1 - b t_{ph}/6)} \tag{7.14}$$

The above equations were inserted into the trajectory model and used to investigate the effects of variable specific impulse operation on a sampling of the NAVIGATOR missions. The use of variable specific impulse, however, introduces two additional degrees of freedom over the constant thrust type of mission. These degrees of freedom can be described by the maximum specific impulse modulation (or equivalently, the thrust or acceleration modulation) and the ratio of propulsion time to trip time. The effect of each of these additional degrees of freedom was investigated in a series of preliminary calculations and found to be relatively unimportant. Final performance is therefore based upon the use of the same propulsion time-trip time relationships obtained for the constant thrust case and upon an acceleration modulation of 40 percent. This last value tends to minimize the specific impulse modulation required for the NAVIGATOR trajectories at about the 10 to 15 percent level.

Figure 7-1 summarizes the results of the variable specific impulse calculation. Payload versus trip time characteristics are shown for both variable and constant specific impulse operation for a number of missions. Although performance advantages of the order of 10 to 15 percent were obtained for the Jupiter I mission, the advantages decreased with the more difficult missions. Some advantage was also identified for the Saturn fly-by mission, the only one investigated.

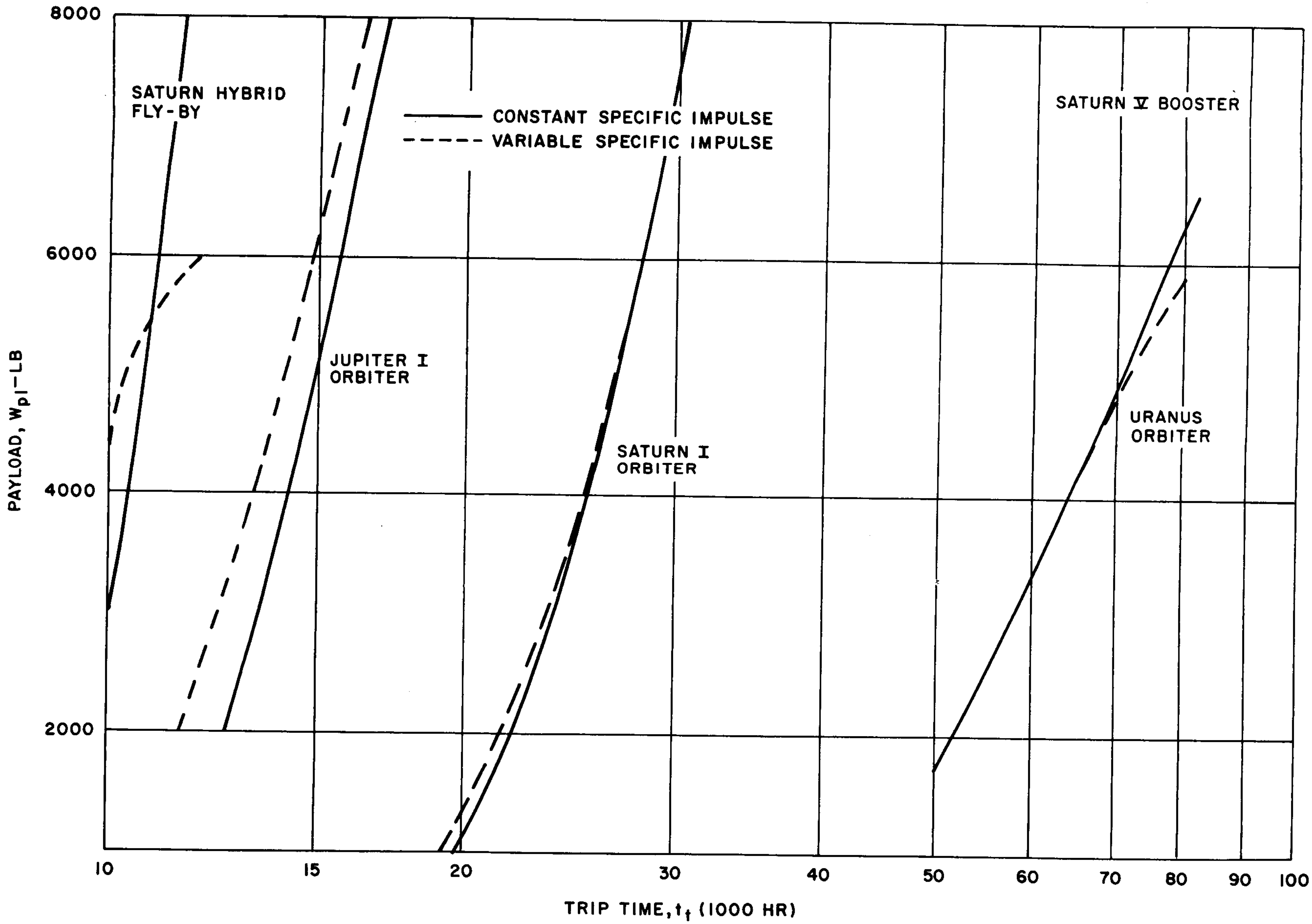


Figure 7-1. Variable Specific Impulse Performance - 30 Pounds Per KW Powerplant

It appears, therefore, that there is a potential performance improvement associated with the use of variable specific impulse for some of the NAVIGATOR missions. Additional investigation, however, will be required to determine whether this performance advantage is sufficient to off-set the system complexities associated with variable specific impulse operation.

8. HIGH THRUST MISSION PERFORMANCE

High thrust trajectory studies were conducted to establish the minimum characteristic velocity requirements for each of the NAVIGATOR orbiter and fly-by missions as a function of trip time. The resulting velocity requirements are used in conjunction with the high thrust performance characteristics of Section 4 to generate mission performance data in terms of payload trip time characteristics. These data, when corrected to equivalent electric propulsion payloads, can be compared with the nuclear-electric performance data of Section 5 to determine either the payload difference at constant trip time or the trip time difference at constant payload. The substantially higher payload power available with the nuclear-electric vehicles, however, suggests that such comparisons should be limited to an assessment of the power and payload differences available at constant trip time.

8.1 TRAJECTORY REQUIREMENTS

The characteristic velocity requirements for the outbound orbiter missions were obtained from the two impulse transfer trajectory illustrated in Figure 8.1-1. The first impulse is added tangentially to the initial circular orbit velocity in the earth's gravitational field to produce a departure hyperbola. The departure velocity is thereafter decreased by the earth's attraction until the vehicle is effectively at infinite distance from the earth (a distance of about 10^6 miles). The resulting hyperbolic excess velocity can be determined from the equation

$$V_{hI} = \sqrt{(V_o + \Delta V_1)^2 - 2 V_o^2} \tag{8.1}$$

The departure hyperbola is assumed to be oriented so that the hyperbolic excess velocity is oriented tangentially with respect to the earth's heliocentric velocity vector.

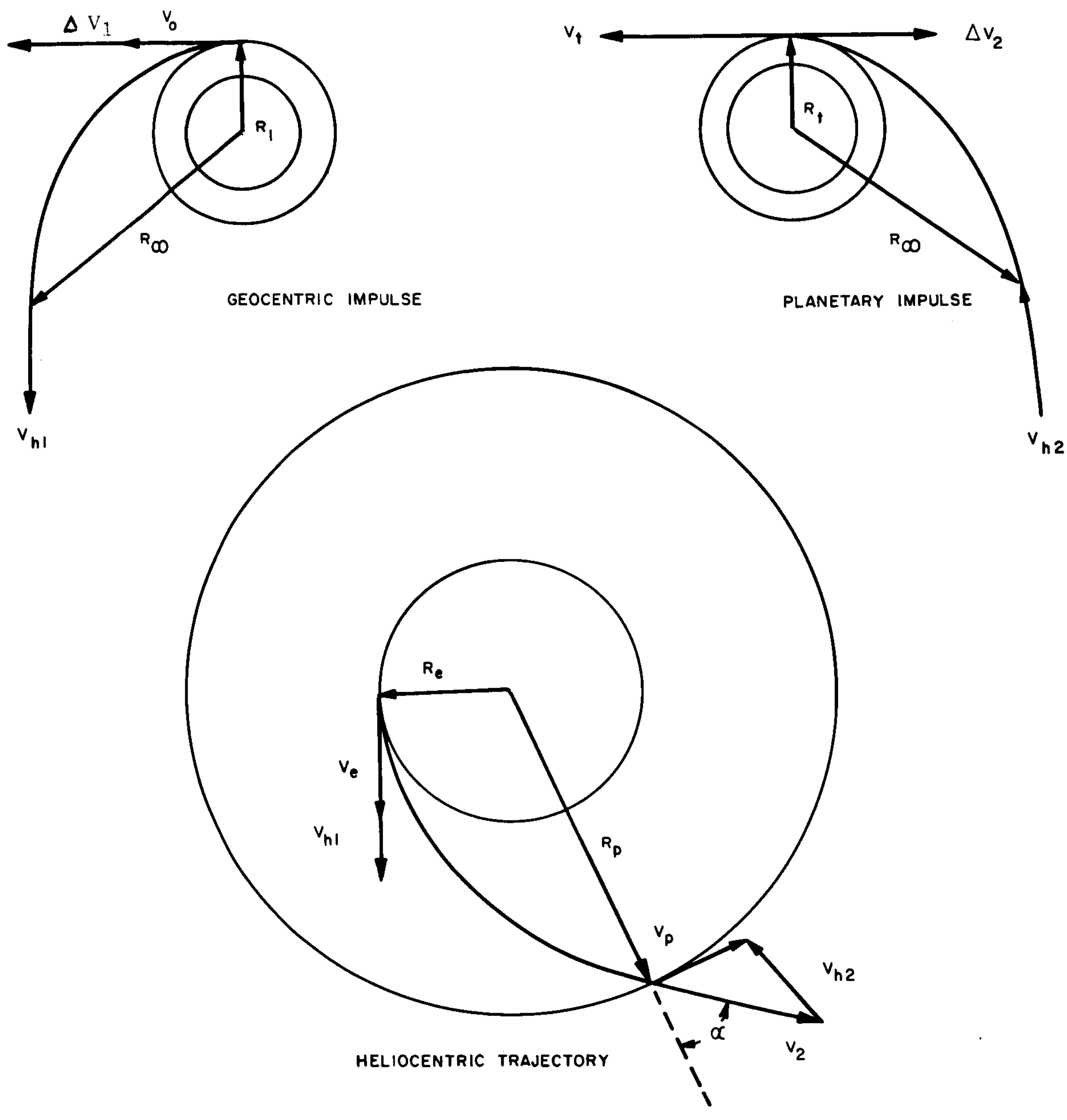


Figure 8.1-1. Two Impulse Transfer Trajectory

The resulting heliocentric departure and arrival velocities can be determined from the equations

$$V_1 = V_e + V_{h1} \tag{8.2}$$

$$V_2 = \sqrt{V_1^{\;2} - 2\,V_e^2\,(1 - \frac{R_e}{R_p})} \quad \text{at}\,\infty = \sin^{-1}\left[\frac{R_e\,V_1}{R_p\,V_2}\right] \tag{8.3}$$

The arrival velocity will result in an approach hyperbolic excess velocity with respect to the target planet which is the vector difference between the heliocentric arrival velocity and the target planet's heliocentric velocity.

$$V_{h2} = V_2 \longrightarrow V_p \tag{8.4}$$

The target planet will then accelerate the arrival velocity as the vehicle approaches the desired terminal orbit altitude. The second tangential impulse will then reduce the approach hyperbola to the desired terminal circular orbit. The magnitude of this impulse is

$$\Delta V_2 = \sqrt{V_{h2}^{\;\;2} + 2\,V_t^{\;2}} - V_t \tag{8.5}$$

The elapsed trip time is assumed to be essentially equal to the heliocentric trip time and is obtained from the conventional Kepler equation (Reference 5).

The inbound orbiter missions are handled in a similar fashion except that the hyperbolic excess velocity with repect to the target planet is oriented tangentially and the earth hyperbolic velocity direction is allowed to vary. This approach complies with the assumed criteria that the inner impulse be tangential. The fly-by velocity requirements are obtained directly from the geocentric impulse of the orbiter trajectories.

Figures 8.1-2 and 8.1-3 summarize the results of these investigations. Figure 8.1-2 contains the characteristic velocity requirements for the major planet orbiter and fly-by missions. The solid lines show the orbiter requirements and the dotted lines, the fly-by

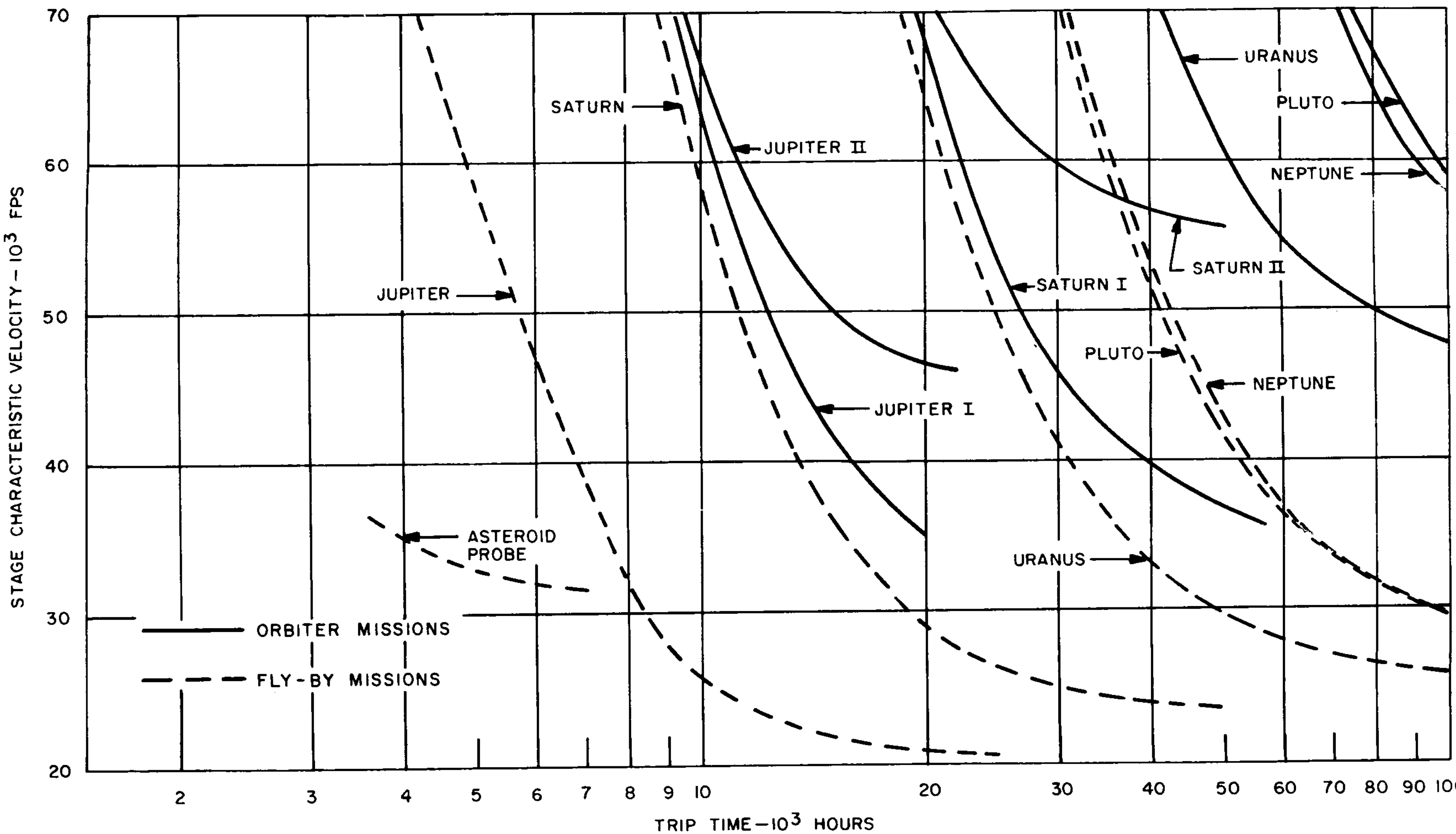


Figure 8.1-2. High Thrust Mission Requirements - Major Planets and Asteroid Probe

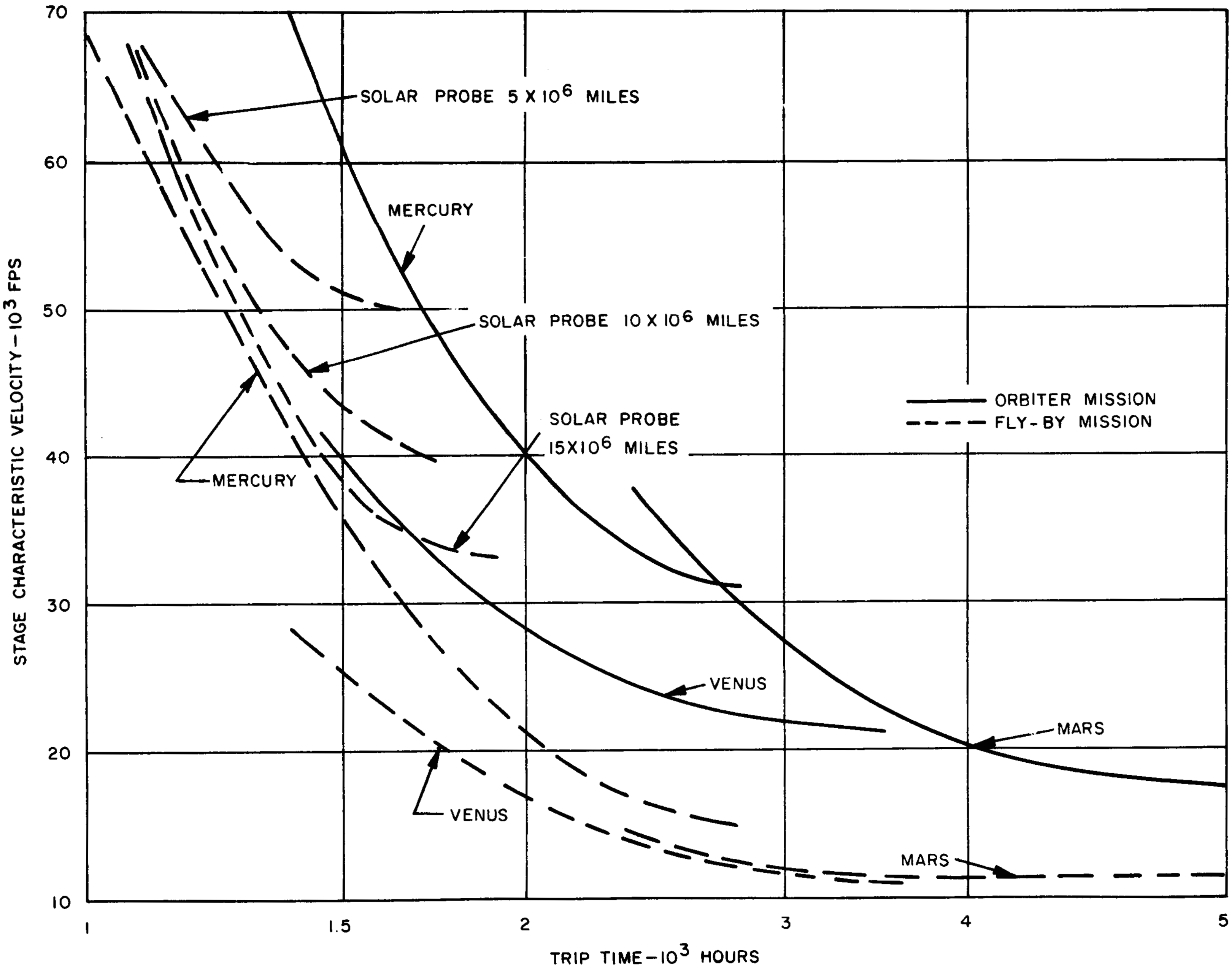


Figure 8.1-3. High Thrust Mission Requirements - Minor Planets and Solar Probes

requirements. These data are limited by the characteristic velocity capabilities of the Saturn boosters at low trip times and by either a Hohman transfer or a trip time limit of 100,000 hours at long trip times. The planetary rendezvous conditions and the terminal planetary orbit altitudes used are identical to those employed in the low acceleration studies of Section 4. Figure 8.1-3 contains comparable data for the minor planet orbiter and fly-by missions. Also shown are the velocity requirements for solar probes which approach to within 5, 10, and 15 million miles of the sun.

The velocity requirement for the out-of-the-ecliptic mission is not shown on the preceeding curves since it is independent of trip time. A 35 degree out-of-the-ecliptic mission will require a characteristic velocity of 43,500 fps and a trip time of 2200 hours to reach maximum declination.

8.2 PERFORMANCE CAPABILITIES

The data of the previous section are combined with the data of Section 6 to obtain payload trip time characteristics for each of the NAVIGATOR missions. Figures 8.2-1 through 8.2-4 summarize the results of these calculations. Figure 8.2-1 contains a summary of the high thrust propulsion performance for the probe and minor planet and Figure 8.2-2 for the major planet fly-by missions. Results are shown for both the Saturn IB booster (Vehicle No. 1) and the Saturn V booster (Vehicle No. 3). The Saturn IB performance appears to be sufficient for the Mercury, Venus, and Mars fly-bys and the Asteriod Probe. The Saturn V provides gross payloads of 10,000 lb or more for all of the missions except the solar probe and out-of-the-ecliptic missions.

Figures 8.2-3 and 8.2-4 contain the comparable results for the orbiter missions. Results, in this case, are illustrated for the Saturn V (Vehicle No. 3) and the Saturn V Nuclear (Vehicle No. 4) boosters. For this case, the Saturn V performance appears to be sufficient for the minor planet orbiters - Mercury, Venus, and Mars - and possibly for limited use with the Jupiter I and Saturn I missions. The Saturn V Nuclear appears to be required to achieve any finite payload for the Jupiter II and Saturn II missions. On the other hand, even the Saturn V Nuclear appears to be insufficient for the Uranus, Neptune, and Pluto orbiter missions.

These data are presented in order to permit comparisons between the high thrust performance capabilities and the low thrust systems described in Section 5. No direct comparisions between the two have been made in this Volume, however, because of the difference in payload value. It is obvious, for example, that a 3,000 pound payload aboard an electric propulsion vehicle that includes a several hundred kilowatt power system has significantly greater value than a 3,000 pound payload aboard a chemical propulsion vehicle. The gross payloads given in Figures 8.2-1 through 8.2-4 are discussed in Volume 3 and compared to equivalent electric propulsion vehicle payloads.

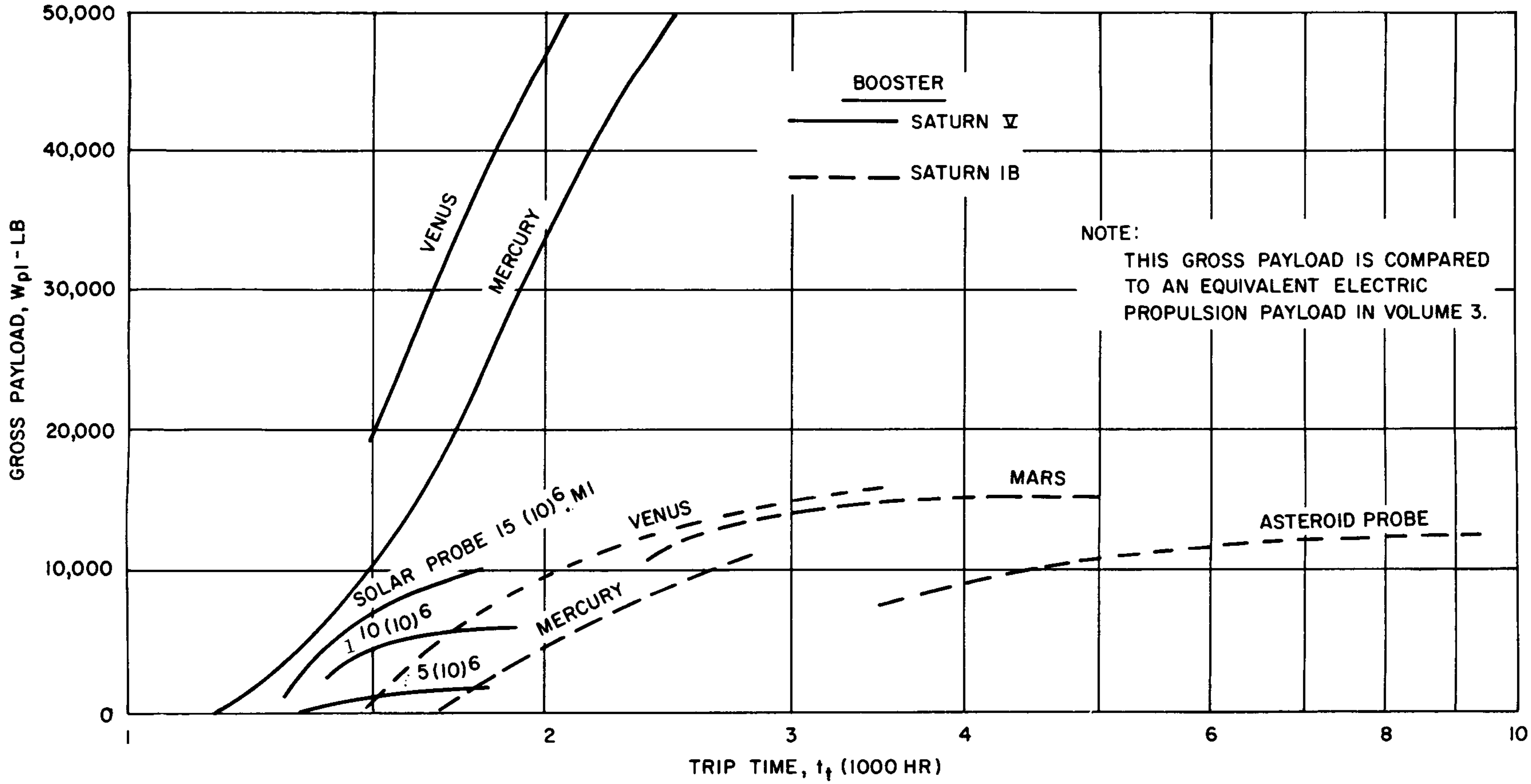


Figure 8.2-1. Probe and Minor Planet Fly-By Performance Summary - High Thrust Propulsion

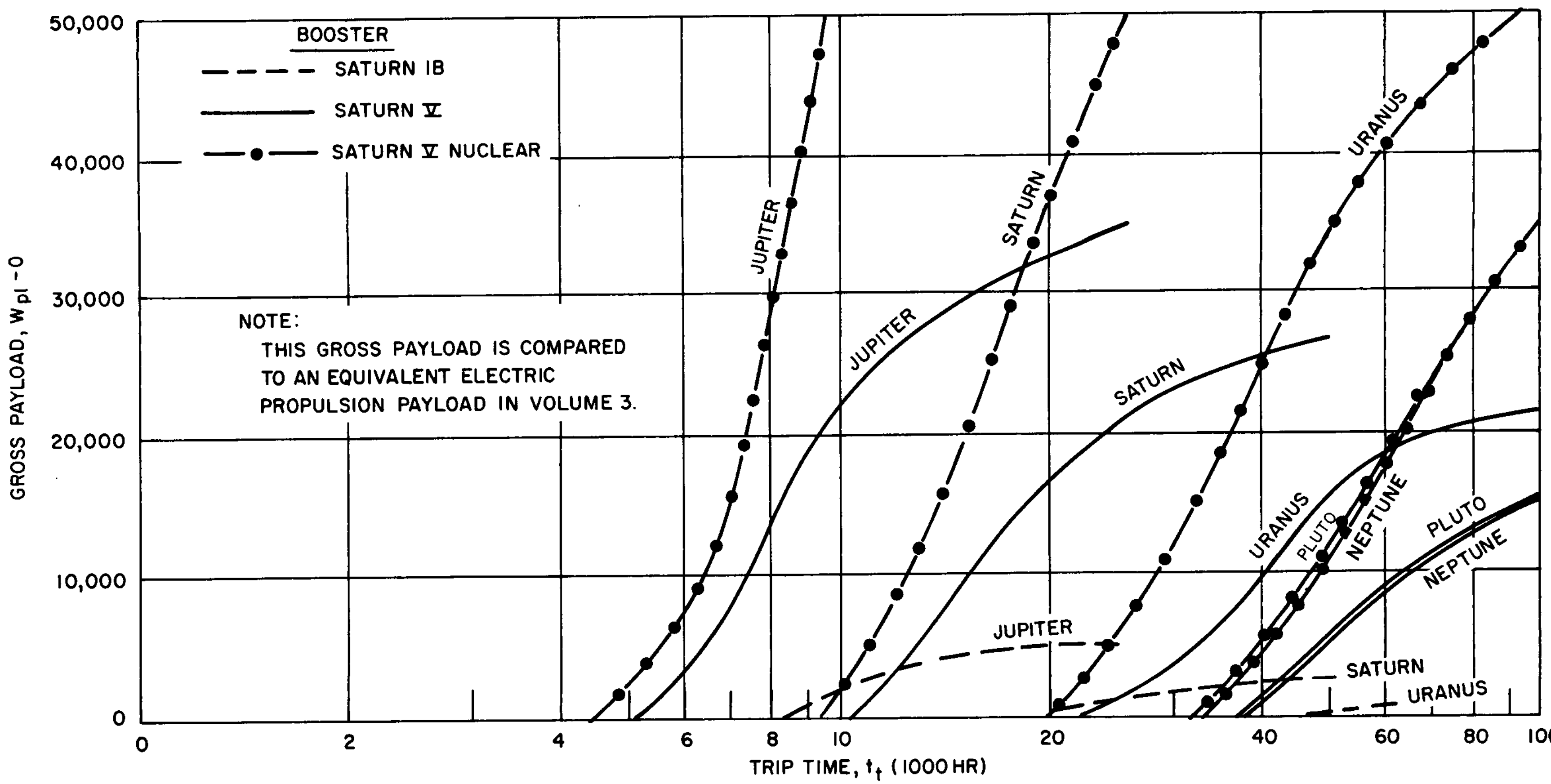


Figure 8.2-2. Major Planet Fly-By Performance Summary - High Thrust Propulsion

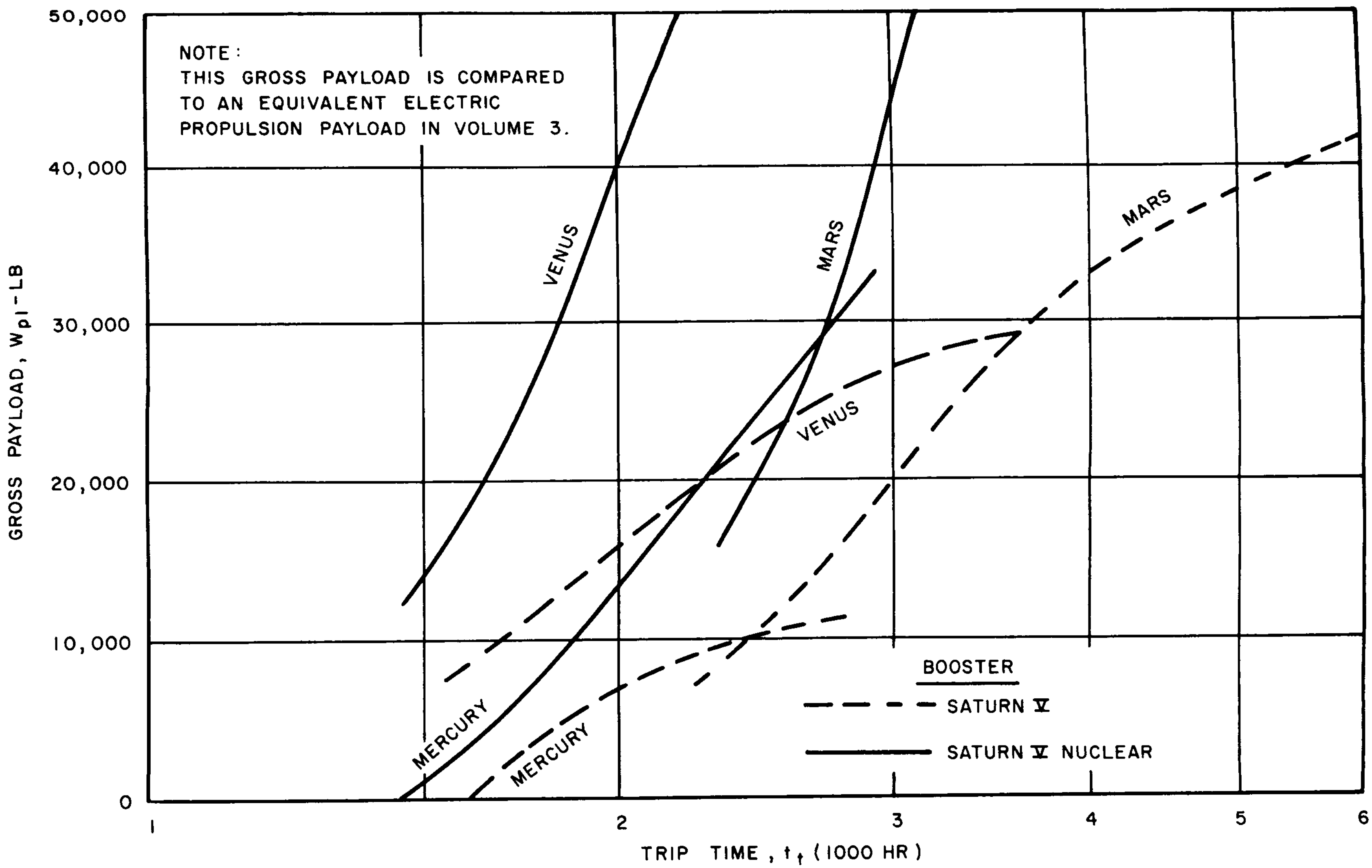


Figure 8.2-3 Minor Planet Orbiter Performance Summary - High Thrust Propulsion

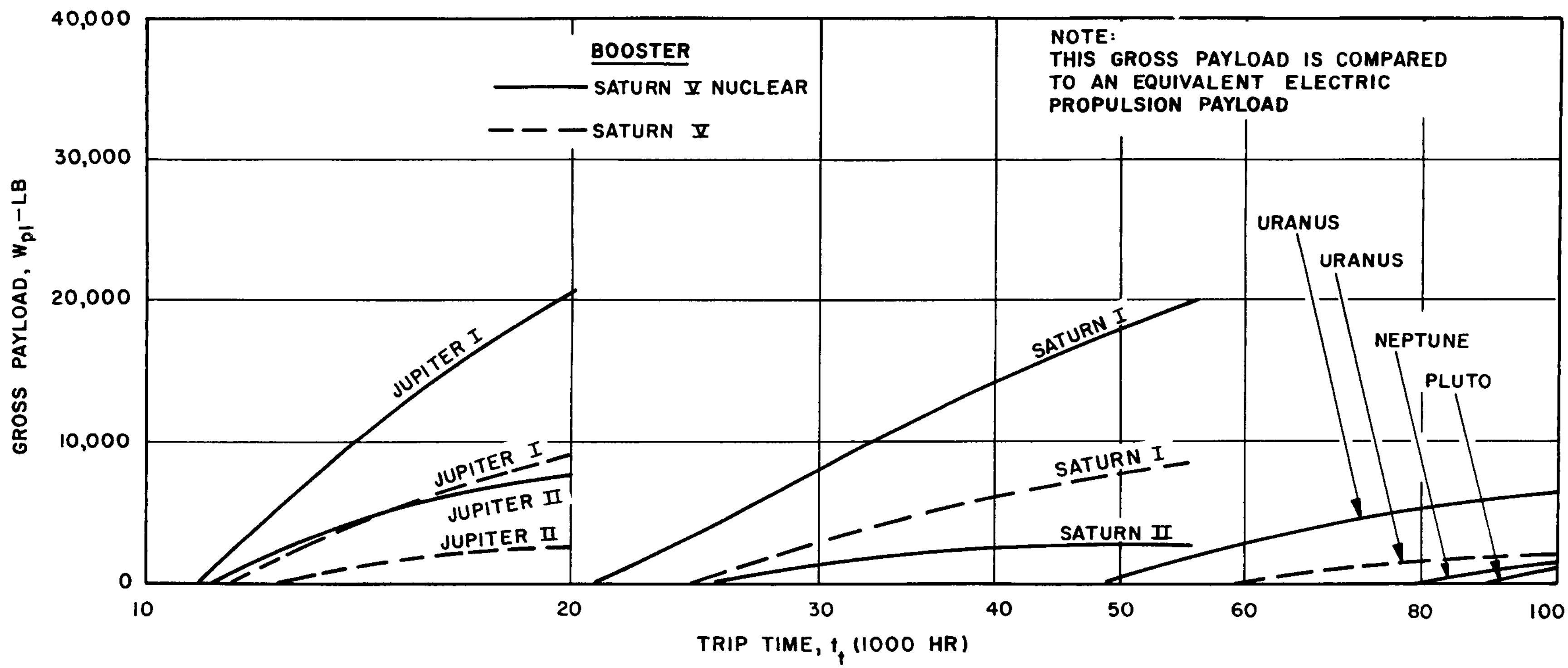


Figure 8.2-4. Major Planet Orbiter Performance Summary - High Thrust Propulsion

9. NOMENCLATURE

a — Low thrust acceleration, miles/hr^2.

a_o — Initial low thrust acceleration, miles/hr^2.

A_o — Coefficient of specific power equation, kw/lb thrust.

A_1 — Coefficient of specific power equation, kw sec/lb thrust.

AU — Astronomical unit, solar distance divided by the mean distance of the Earth from the Sun.

Constant thrust-optimum coast, low acceleration heliocentric trajectory optimized to minimize J with constant thrust operation. Results in intermediate coast period.

Declination, celestial lattitude measured with respect to the ecliptic plane.

Ecliptic plane, the plane of the Earth's orbit about the Sun.

Fly-by trajectory, one which matches position but not velocity with target planet.

g — Sea level gravitational acceleration, 79,019 miles/hr^2.

G — Universal gravitational constant, 9.40382 $(10)^{14}$ miles^3/lb hr^2.

Geocentric, central body motion with the Earth as the center of the force field.

Heliocentric, central body motion with the Sun as the center of the force field.

High thrust, acceleration involving thrust weight ratios greater than $(10)^{-1}$.

Hyperbolic excess velocity, the geocentric or planetary residual velocity at infinite distance from the center of the force field.

I — Inclination angle, the angle between an orbit plane and the ecliptic plane.

I_h — Inclination angle change generated by high thrust.

I_1 — Inclination angle change generated by low thrust.

I_{sp} — Thruster specific impulse, lb thrust/lb per second fuel, seconds.

J Low acceleration propulsion parameter, miles^2/hr^3.

L Characteristic length, measure of low acceleration propulsion requirements, miles.

L_m Minimum characteristic length, miles.

L_o Characteristic length parameter extrapolated to zero trip time, miles.

Low thrust, acceleration involving thrust weight ratios less than $(10)^{-3}$.

M_e Mass of the Earth, 1.3177 $(10)^{25}$ lb.

M_p Mass of the target planet.

M_s Mass of the Sun, 4.3894 $(10)^{30}$ lb.

N Vector normal to orbital plane.

Optimum variable specific impulse, low acceleration heliocentric trajectory optimized to minimize J at constant power. Results in large (40:1) specific impulse variation.

Orbital period, the period of revolution of an orbit.

Orbital plane, the plane defined by the instantaneous radius and velocity vectors with respect to the central body.

Orbiter trajectory, one which matches both position and velocity with the target planet and which can be converted to a low altitude planetary orbit with additional propulsion.

P Power rating, kw.

P_o Radius of orbit with respect to Earth, miles.

P_f Radius of orbit with respect to target planet, miles.

Perihelion, the point on a heliocentric orbit which is closest to the Sun.

Planetary, central body motion with the target planet as the center of the force field.

Quasi-circular, an orbit approximation in which the actual velocity is assumed to be identical with the circular orbital velocity.

R Radius vector with respect to the Sun, miles.

R_e Radius of the Earth's orbit with respect to the Sun, miles.

R_o Radius with respect to the Earth, miles.

R_p Radius of the target planet with respect to the Sun, miles.

R_t Radius with respect to the target planet, miles.

R_∞ The equivalent of infinite radius at which the Earth or planet no longer has any effect on the orbit.

t Time, hr.

t_c Coast time, hr.

t_h Heliocentric trip time, hr.

t_m Trip time at which characteristic length minimizes, hr.

t_p Low acceleration propulsion time, hr.

t_{ph} Heliocentric propulsion time, hr.

t_{pl} Planetocentric propulsion time, hr.

t_t Total trip time, hr.

T Thrust, lb.

Two point boundary problem, problem involving a number of constraints at the initial and terminal ends of a trajectory which must be solved iteratively to satisfy the terminal conditions.

V One dimensional velocity obtained by integrating acceleration in field free space or heliocentric velocity vector.

V_e Velocity of the Earth with respect to the Sun, mph.

V_{h1} Hyperbolic excess velocity with respect to the Earth, mph.

V_{h2} Hyperbolic excess velocity with respect to the target planet, mph.

V_j Thruster jet velocity, mph.

V_o	Initial orbital velocity with respect to Earth, mph.
V_o	Initial one dimensional velocity and equal to V_{h1}.
V_t	Terminal orbit velocity with respect to planet, mph.
V_2	One dimensional velocity at coast, mph.
V_3	Terminal one dimensional velocity and equal to V_{h2}.
ΔV	Low thrust characteristic velocity and equal to $g\ I_{sp}\ 1\ n\mu$, mph.
ΔV_{cl}	Constant low thrust heliocentric characteristic velocity, mph.
ΔV_g	Geocentric ΔV requirement for achieving parabolic escape from initial circular orbit at 300 miles, mph.
ΔV_h	Heliocentric characteristic velocity requirement, mph.

10. REFERENCES

1. H. Brown and T. F. Widmer. "Research on Spacecraft and Powerplant Integration Problems", Mission Analysis Topical Report - Contract No. NAS 3-2533, G. E. Technical Report No. 64SD505. Philadelphia, 1964.

2. W.G. Melbourne. "Interplanetary Trajectories and Payload Capabilities of Advanced Propulsion Systems", JPL Technical Report No. 32-68, 1961.

3. C. L. Zola. "Trajectory Methods in Mission Analysis for Low-Thrust Vehicles", NASA-Lewis Technical Preprint No. 20-63, 1963.

4. T. N. Edelbaum. "Propulsion Requirements for Controllable Satellites", ARS Journal, Vol. 31, No. 8, 1961.

5. F. R. Moulton. An Introduction to Celestial Mechanics. Second Edition. The MacMillan Company. New York, 1914.